NACA

RESEARCH MEMORANDUM

ALTITUDE-WIND-TUNNEL INVESTIGATION OF A 3000-POUND-THRUST AXIAL-FLOW TURBOJET ENGINE

VII - PRESSURE AND TEMPERATURE DISTRIBUTIONS

By Martin J. Saari and William R. Prince

Flight Propulsion Research Laboratory
Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON
December 10, 1948

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

ALTITUDE-WIND-TUNNEL INVESTIGATION OF A 3000-POUND-THRUST AXIAL-FLOW TURBOJET ENGINE

VII - PRESSURE AND TEMPERATURE DISTRIBUTIONS

By Martin J. Saari and William R. Prince

SUMMARY

An investigation has been conducted in the NACA Cleveland altitude wind tunnel during which temperature and pressure distributions throughout an original and a modified 3000-pound-thrust turbojet engine were obtained. Data are presented for both engines for a range of simulated altitudes from 5000 to 45,000 feet, simulated flight Mach numbers from 0.24 to 1.08, and corrected engine speeds from 10,550 to 13,359 rpm. The effects of altitude, flight Mach number, and corrected engine speed on the pressure and temperature distributions at each measuring station of both engines are discussed.

Modification of the compressor, the combustion chamber, and the fuel-spray nozzles of the original engine improved the radial total-pressure distribution at the compressor outlet and the radial and circumferential temperature distribution at the turbine outlet. Pressure and temperature distributions at all measuring stations through the engines, except the compressor outlet and the turbine outlet, were not appreciably affected by changes in altitude, flight Mach number, or engine speed. An increase in altitude from 25,000 to 45,000 feet resulted in high total-pressure peaks near the outer wall of the compressor outlet. For the original engine, this increase in altitude resulted in an increase of temperature near the inner wall of the turbine outlet. For both engines, an increase in engine speed moved the temperature peaks at the turbine outlet toward the root sections of the turbine blades. Except for slight differences in pressure levels, variations of average total and static pressures throughout the original and modified engines were similar for comparable operating conditions.

INTRODUCTION

An investigation has been conducted in the NACA Cleveland altitude wind tunnel to determine the performance and operational characteristics of original and modified configurations of a 3000-pound-thrust axial-flow turbojet engine. In the initial phase of the investigation of the original engine, the engine was operated with an exhaust-nozzle-outlet average temperature limit of 1210° F. Operation of the engine at this temperature limit resulted in a failure of the turbine assembly. It was found that the tail-rake temperature measurements were not representative of the temperature to which the turbine was subjected, owing to undesirable circumferential and radial temperature variations at the turbine inlet. The temperature limit was then changed to 1375° F, as read on the hottest thermocouple at the turbine outlet. Operation at the revised temperature limit resulted in a second failure of the turbine rotor blades. It was found that the existing thermocouples around the circumference of the turbine outlet were not located in regions of maximum temperature. Additional instrumentation was installed at the turbine outlet and the temperature limit was lowered to 1250° F, as read on the hottest thermocouple. The investigation of the original engine was completed with the revised temperature limit without further turbine failure.

In order to improve the turbine-inlet temperature distribution and thereby increase the allowable turbine-outlet temperature limit, the compressor, the combustion chamber, and the fuel-spray nozzles were modified for the second part of the investigation. The operating temperature of the modified engine was increased to 1400° F, as indicated by the hottest thermocouple at the turbine outlet.

The investigation was conducted over a wide range of simulated altitudes, simulated flight Mach numbers, and engine speeds. Analyses of turbine performance, compressor performance, combustion-chamber performance, operational characteristics, over-all engine performance, and inlet pressure losses are presented in references 1 to 6, respectively.

Data are presented herein to show the effect of altitude, flight Mach number, and engine speed on pressure and temperature distributions at each measuring station within the original and modified configurations. The data presented for both engines have been generalized to NACA standard sea-level conditions.

944

DESCRIPTION OF ENGINES

The X24C-4B turbojet engine used in this investigation has a static sea-level thrust rating of 3000 pounds at an engine speed of 12,500 rpm. At this rating the air flow is approximately 58.5 pounds per second and the fuel consumption is 3200 pounds per hour. The over-all length of the engine is $119\frac{1}{2}$ inches, the maximum diameter is $28\frac{1}{4}$ inches, and the total weight is 1150 pounds. The main components of the engine include an 11-stage axial-flow compressor, a double-annulus combustion chamber, a two-stage turbine, and a fixed-area exhaust nozzle.

The main components of the two engines used in this investigation were similar except for modifications made to the compressor and the combustion chamber by the manufacturer. The compressor was modified to improve the radial velocity distribution at the compressor outlet by twisting the eleventh-stage rotor blades, in the direction of reduced angle of attack, 3° at the midspan and 6° at the blade tip.

The combustion chamber was modified to improve the temperature distribution at the turbine inlet by omitting the wall perforations of the fourth step of the combustion-chamber liner and replacing the holes in the third step with a single row of large rectangular holes, the area of which equalled the total area of the third- and fourth-step holes in the original liner. The blocking area of the screens at the combustion-chamber inlet was reduced. For the original configuration, a screen having 60-percent blocking area was installed in the outer annulus and one having 40-percent blocking area was installed in the intermediate annulus. For the modified engine, these screens were replaced by screens having 30-percent blocking area. In order to obtain a more uniform circumferential temperature distribution at the turbine inlet, fuel nozzles of a different type were installed in the modified engine. The fuel nozzles for the original engine had a rated capacity of $7\frac{1}{2}$ gallons per hour at a differential pressure of 100 pounds per square inch, as compared to 7 gallons per hour for the nozzles in the modified engine.

As a result of these modifications, the operating temperature limit, as indicated by the hottest thermocouple at the turbine outlet, was raised from 1250° F for the original engine to 1400° F for the modified engine. A reduction in exhaust-nozzle-outlet area from 183 square inches for the original engine to 171 square inches

for the modified engine was consequently required to obtain the higher temperature limit. More detailed descriptions of the engines and their components are presented in references 1 to 5.

INSTALLATION AND INSTRUMENTATION

The engine was installed in a wing nacelle in the test section of the altitude wind tunnel (fig. 1). Compressor-inlet pressures corresponding to high flight Mach numbers were simulated by introducing dry refrigerated air from the tunnel make-up air system through a duct to the engine inlet. This air was throttled from approximately sea-level pressure to the desired pressure at the compressor inlet while the static pressure in the tunnel was maintained to correspond to the desired altitude. The duct was connected to the engine inlet by means of a slip joint with a labyrinth seal, which permitted engine-thrust and installation-drag measurements to be made with the tunnel balance scales.

Temperature and pressure measurements were obtained at several stations through the engine (fig. 2). Drawings of the instrumentation at the compressor inlet, the compressor outlet, the turbine inlet, the turbine stator stages, the turbine outlet, and the exhaust-nozzle outlet are presented in figures 3 to 8, respectively.

PROCEDURE

Pressures and temperatures throughout the engine are presented for simulated altitudes from 5000 to 45,000 feet, simulated flight Mach numbers from 0.24 to 1.08, and corrected engine speeds from 10,550 to 13,359 rpm. For most operating conditions, the inlet-air temperature was held at approximately NACA standard values corresponding to the simulated flight conditions. No inlet-air temperatures below approximately -20° F, corresponding to high altitude and low flight Mach number, were obtained.

Temperatures were measured and recorded by two self-balancing potentiometers. Pressures were measured by water, alkazene, and mercury manometers and were photographically recorded. The engine speed was set by means of a stroboscopic tachometer.

RESULTS AND DISCUSSION

Data are presented to show the effect of altitude, flight Mach number, and corrected engine speed on the pressure and temperature

944

distribution at each measuring station and on the variation of average pressures throughout the original and modified engines. In order to provide a basis for comparing the engines, the data for both engines have been generalized to standard sea-level conditions by the use of the factors δ (ratio of absolute total pressure at compressor inlet to absolute static pressure corresponding to NACA standard atmosphere at sea level) and θ (ratio of absolute indicated total temperature at compressor inlet to absolute static temperature of NACA standard atmosphere at sea level). Pressure and temperature distributions are presented for both engines at comparable simulated flight conditions. The effect of altitude at a flight Mach number of 0.24 and a corrected engine speed $N/\sqrt{\theta}$ of approximately 12,000 rpm is presented for simulated altitudes from 5000 to 45,000 feet. The effect of flight Mach number at an altitude of 25,000 feet and a corrected engine speed of approximately 12,700 rpm is presented for flight Mach numbers of 0.53 to 1.08. The corrected engine speeds at which the effects of altitude and flight Mach number are shown were the maximum speeds at which comparable data were obtained over the complete range of conditions for both engines. The effect of engine speed at a flight Mach number of 0.53 and an altitude of 25,000 feet is presented for corrected engine speeds from 10,550 to 13,359 rpm.

Compressor Inlet

The radial distribution of total pressure, static pressure, and indicated temperature at the compressor inlet is shown in figure 9 for the original and modified engines. Total- and static-pressure levels for both engines were nearly the same at each operating condition. The maximum variation of temperature levels between the two engines at any operating conditions was about 15° F.

Total-pressure, static-pressure, and indicated-temperature distributions were uniform across the passage for both engines at all operating conditions investigated, with the exception of slightly lower total pressures near the walls due to the boundary layer.

Compressor Stator Stages

The variation of corrected static pressure P_3/δ at the compressor stator stages is shown in figure 10 for the original and modified engines. Continuous pressure rise was obtained throughout the stages of the compressor of both engines for the operating

conditions shown. An increase in altitude beyond 25,000 feet increased the over-all corrected pressure through the compressor. Increasing the flight Mach number from 0.53 to 1.08 did not appreciably affect the general pressure distribution through the first eight compressor stages; for the remaining stages, an increase in flight Mach number reduced the pressure level. For the modified engine (fig. 10(b)), the corrected engine speeds at flight Mach numbers of 0.53 and 1.08 were somewhat lower than at the comparable conditions for the original engine, and consequently, the pressure levels are not in the same sequence with increasing flight Mach number as shown for the original engine. Increasing the corrected engine speed lowered the static pressure at the first-stator stage as a result of the increased velocities through the inlet guide vanes. Twisting the eleventh-stage rotor blades had no appreciable effect on the static-pressure rise across that stage. The effects of altitude, flight Mach number, and engine speed on the corrected static-pressure distributions through the compressors were similar for both engines.

Compressor Outlet

Average radial distributions of total pressure P_4/δ at the compressor outlet are shown in figure 11 for both engines. Inasmuch as no appreciable radial or circumferential variation in static pressure occurred at any operating condition, only the average corrected static pressure p_4/δ is shown.

For the original engine (fig. 11(a)), the corrected total pressure and the average static pressure were approximately equal for a distance of 3/4 inch from the inner wall of the passage at each altitude, which indicated low velocity and air flow. From this point, the corrected total pressure increased rapidly across the passage and, for altitudes up to 25,000 feet, peak total pressures were obtained at approximately 3/4 inch from the outer wall. At an altitude of 45,000 feet, the total-pressure peak increased in magnitude and moved nearer the outer wall. For the modified engine (fig. 11(b)), the total-pressure peak at an altitude of 5000 feet occurred in the central part of the compressor-outlet passage and at an altitude of 45,000 feet the total-pressure peak moved outward to approximately 3/4 inch from the outer wall. The total and static pressures near the inner wall, however, were approximately equal, as in the original engine.

The total-pressure distribution for the original engine was not appreciably affected by changes in flight Mach number except

near the outer wall, where, at the high flight Mach number, the corrected total pressure was less than the peak corrected total pressure. For the modified engine, the total-pressure distributions for flight Mach numbers of 0.53 and 0.87 were similar, with the peak occurring 3/4 inch from the outer wall; however, at a flight Mach number of 1.08 the total-pressure peak occurred in the central part of the passage.

For both engines, the total-pressure peak moved from the center of the passage at the lowest corrected engine speed to a point near the outer wall at the highest corrected engine speed. For the original engine, the peak total pressure at maximum engine speed occurred about 1/4 inch from the outer wall; whereas for the modified engine the total-pressure peak occurred about 3/4 inch from the outer wall.

In general, modifying the compressor shifted the total-pressure peak toward the center of the compressor-outlet passage but did not appreciably affect the pressure distribution near the inner wall.

Turbine Inlet

Circumferential distribution of corrected total pressure P_5/δ at the turbine inlet is shown in figure 12 for the original and modified engines. The total-pressure distribution was uniform for both engines at all operating conditions. Changes in altitude, flight Mach number, or engine speed affected only the pressure level.

Turbine Outlet

Total-pressure distributions at the turbine outlet are presented in figure 13 for both engines. The static pressure shown for each operating condition is an average of three static-wall-orifice measurements. In general, the corrected total pressure P_7/δ increased across the passage from the inner wall to the outer wall. For the modified engine (fig. 13(b)), the rapid increase in total pressure near the outer wall is attributed to the reduced blocking area of the screen at the inlet of the outer annulus of the combustion chamber. The distribution for both engines was not greatly affected by changes in operating conditions.

The radial distribution of corrected turbine-outlet temperature $T_{i,7}/\theta$ at three circumferential positions for altitudes

from 5000 to 45,000 feet is presented in figure 14 for the original and modified engines at a flight Mach number of 0.24 and a corrected engine speed of about 12,000 rpm. The turbine-outlet temperature distribution for the original engine (fig. 14(a)) was not affected, except near the outer wall, when the altitude was increased from 5000 to 25,000 feet. At an altitude of 45,000 feet, the temperature at the inner wall increased with respect to the temperature at the outer wall to such an extent that the highest temperatures occurred in the vicinity of the turbine-blade roots. For the modified engine (fig. 14(b)), the temperature distributions measured by each rake were similar for the range of altitudes investigated. For both engines, an increase in altitude from 5000 to 25,000 feet generally resulted in a small increase in the corrected temperature level. Increasing the altitude beyond 25,000 feet, however, considerably increased the temperature level.

The radial distribution of corrected temperature at the turbine outlet for flight Mach numbers from 0.53 to 1.08 is presented in figure 15 for both engines at an altitude of 25,000 feet and a corrected engine speed of about 12,700 rpm. For both engines, changes in flight Mach number did not affect the temperature distribution. An increase in flight Mach number decreased the corrected temperature level except at a flight Mach number of 0.87 for the modified engine. The corrected engine speeds at flight Mach numbers of 0.53 and 1.08 for the modified engine were somewhat lower than at the comparable conditions for the original engine and, consequently, the temperature levels are not in the same sequence with increasing flight Mach number as shown for the original engine.

The effect of corrected engine speed on the radial distribution of corrected indicated temperature at the turbine outlet is shown in figure 16 for both engines at an altitude of 25,000 feet and a flight Mach number of 0.53. Increasing the corrected engine speed moved the temperature peak toward the root sections of the turbine blades. However, this effect was more pronounced for the original engine than for the modified engine. At the highest engine speed for the modified engine (fig. 16(b)), the maximum temperature at each rake position occurred in the central part of the passage.

The effects of altitude, flight Mach number, and corrected engine speed on the average radial turbine-outlet temperature distribution are shown in figure 17 for both engines. The maximum variation of average temperature across the turbine-outlet passage at any flight condition for the original engine was about 160° F. The high temperatures that occurred near the inner wall at high

altitude and high engine speed for the original engine were undesirable from turbine-blade stress considerations. With regard to blade stress, it is desirable to have the lowest temperatures near the blade roots and the peak temperature near the blade tips. For the modified engine (fig. 17(b)), a temperature peak existed in the central part of the passage for all operating conditions. The temperature distribution for the modified engine corresponded more closely to the aforementioned design temperature distribution.

The circumferential distribution of corrected indicated temperature at a distance of $1\frac{3}{4}$ inches from the tail-pipe outer wall is shown in figure 18 at each operating condition for the original and modified engines. Values for the circumferential distribution were obtained from the three rakes and the individual thermocouples shown in figure 7. Large circumferential temperature variations occurred at all operating conditions for both engines. An inspection of the turbine-stator-blade assemblies showed discoloration around the circumference, which indicate the presence of the temperature irregularities shown at the turbine outlet. Furthermore, it is believed that the distribution at the turbine inlet was even more irregular than indicated by turbine-outlet measurements, inasmuch as the magnitude of the temperature variations was reduced by mixing in passing through the turbine rotors. The circumferential temperature distributions of both engines were not greatly affected by a change in altitude from 5000 to 25,000 feet. An increase in altitude to 45,000 feet raised the temperature level and resulted in somewhat higher temperatures in the lower part of the turbine outlet than in the upper part. This temperature difference was probably caused by unequal flow from the upper and lower fuel nozzles. The difference in head of fuel between the top and the bottom of the fuel manifold, together with the low fuel-manifold pressure accompanying high-altitude operation, resulted in decreased fuel flow and poor atomization from the top nozzles.

Changes in flight Mach number had no effect on the circumferential distribution of corrected indicated temperature at the turbine outlet of either engine. Except for the change in temperature level, variation of corrected engine speed had no appreciable effect on the circumferential temperature distribution of either engine.

Modification of the original engine resulted in a more uniform temperature distribution over a greater part of the turbine-outlet circumference, but did not reduce the peak temperatures.

Exhaust-Nozzle Outlet

The effect of altitude, flight Mach number, and corrected engine speed on the distribution of corrected total pressure P_8/δ, static pressure p_8/δ, and indicated temperature $T_{i,8}/\theta$, at the exhaust-nozzle outlet is shown in figure 19 for the original and modified engines. The corrected total-pressure distribution was reasonably symmetrical about the center line of the jet at all operating conditions for both engines. In all cases, the total pressures at the center of the jet were lower than at the wall of the exhaust nozzle. The corrected temperature was reasonably uniform across the upper half of the exhaust nozzle at all flight conditions, except at an altitude of 45,000 feet. At 45,000 feet, the decreased temperature in the upper part of the exhaust nozzle relative to the center of the jet can be attributed to poor fuel distribution at high altitudes.

Engine Profile

The variations of average total and static pressures through the original and modified engines for all operating conditions previously discussed are shown in figures 20 to 22. Except for slight differences in pressure levels, the over-all pressure distributions for the original and modified engines were similar at comparable operating conditions.

SUMMARY OF RESULTS

The following results were obtained from an investigation of original and modified configurations of a 3000-pound-thrust axial-flow turbojet engine in the Cleveland altitude wind tunnel at simulated altitudes from 5000 to 45,000 feet, simulated flight Mach numbers from 0.24 to 1.08, and corrected engine speeds from 10,550 to 13,359 rpm:

1. Modification of the compressor, the combustion chamber, and the fuel nozzles of the original engine improved the radial total-pressure distribution at the compressor outlet and the radial and circumferential temperature distributions at the turbine outlet.

2. Pressure and temperature distributions at all measuring stations, except the compressor outlet and the turbine outlet, were not greatly affected by changes in altitude, flight Mach number, or engine speed. An increase in altitude from 25,000 to 45,000 feet

944

resulted in high total-pressure peaks near the outer wall of the compressor-outlet passage of both engines. For the original engine, this increase in altitude resulted in an increase of temperatures near the inner wall of the turbine outlet. Increasing the corrected engine speed moved the temperature peaks at the turbine outlet toward the root sections of the turbine blades. This effect was more pronounced for the original engine than for the modified engine.

3. With the exception of slight differences in pressure levels, the variations of average total and static pressures throughout the original and modified engines were similar for comparable operating conditions.

Flight Propulsion Research Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio.

REFERENCES

1. Conrad, Earl W., Dietz, Robert O., Jr., and Golladay, Richard L.: Altitude-Wind-Tunnel Investigation of a 3000-Pound-Thrust Axial-Flow Turbojet Engine. I - Analysis of Turbine Performance. NACA RM No. E8A23, 1948.

2. Dietz, Robert O., Jr., Berdysz, Joseph J., and Howard, Ephraim M.: Altitude-Wind-Tunnel Investigation of a 3000-Pound-Thrust Axial-Flow Turbojet Engine. II - Analysis of Compressor Performance. NACA RM No. E8A26a, 1948.

3. Campbell, Carl E.: Altitude-Wind-Tunnel Investigation of a 3000-Pound-Thrust Axial-Flow Turbojet Engine. III - Analysis of Combustion-Chamber Performance. NACA RM No. E8B19, 1948.

4. Hawkins, W. Kent, and Meyer, Carl L.: Altitude-Wind-Tunnel Investigation of Operational Characteristics of Westinghouse X24C-4B Axial-Flow Turbojet Engine. NACA RM No. E8J25, 1948.

5. Meyer, Carl L., and Bloomer, Harry E.: Altitude-Wind-Tunnel Investigation of Performance and Windmilling Drag Characteristics of Westinghouse X24C-4B Axial-Flow Turbojet Engine. NACA RM No. E8J25a, 1948.

6. Sanders, Newell D., and Palasics, John: Analysis of Effects of Inlet Pressure Losses on Performance of Axial-Flow Type Turbojet Engine. NACA RM No. E8J25b, 1948.

944

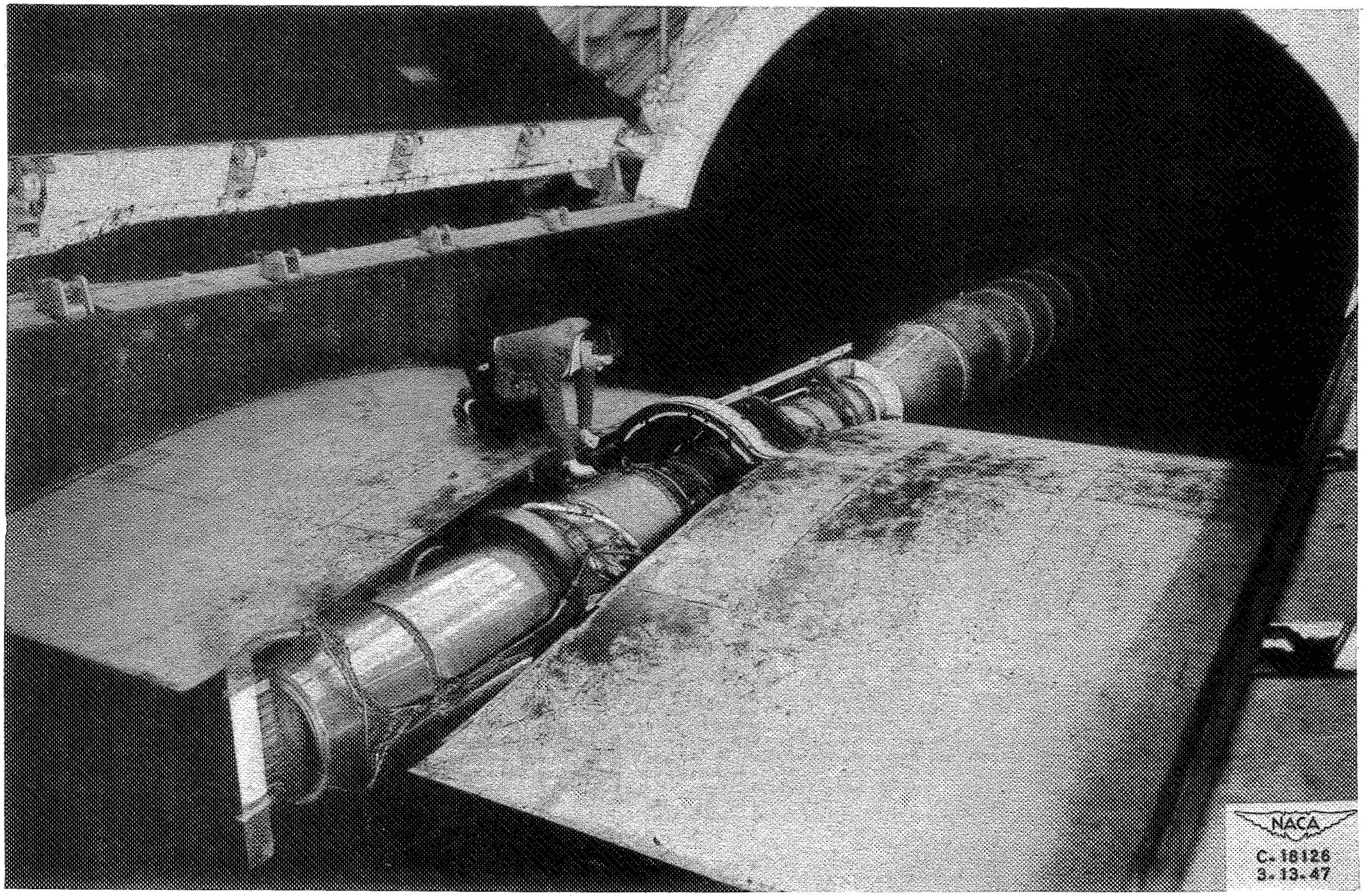


Figure 1. - Installation of turbojet engine in altitude wind tunnel.

CONFIDENTIAL

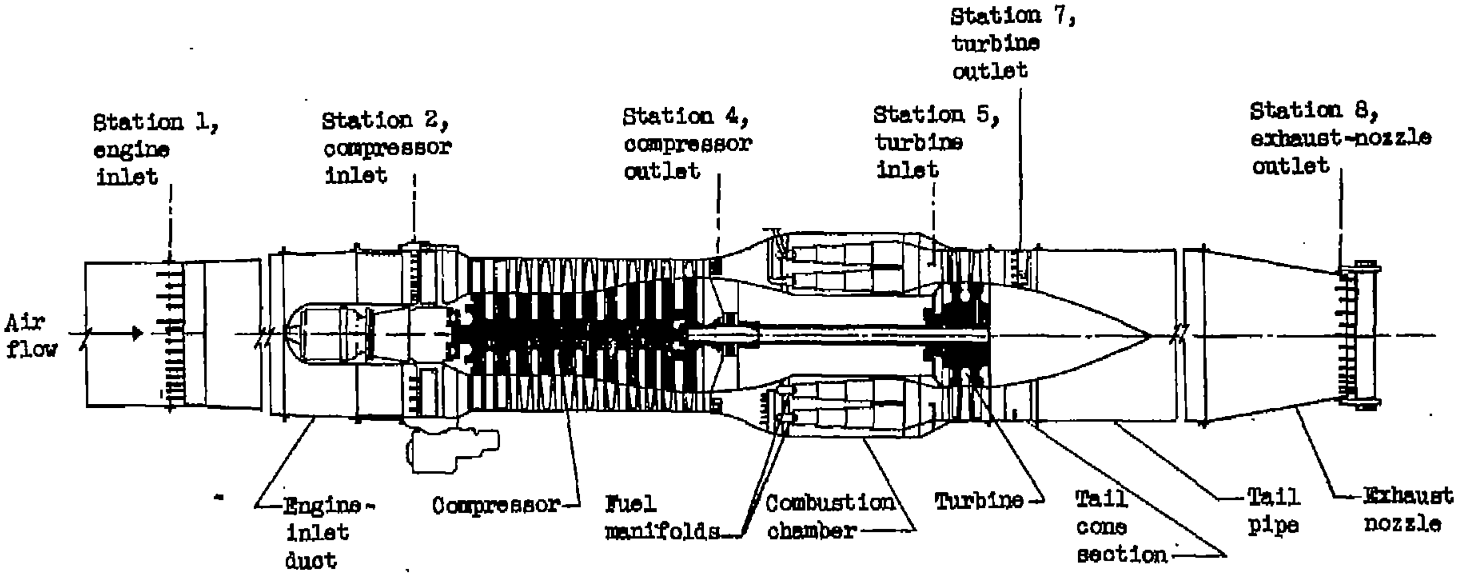


Figure 2. - Location of instrumentation installed in turbojet engine.

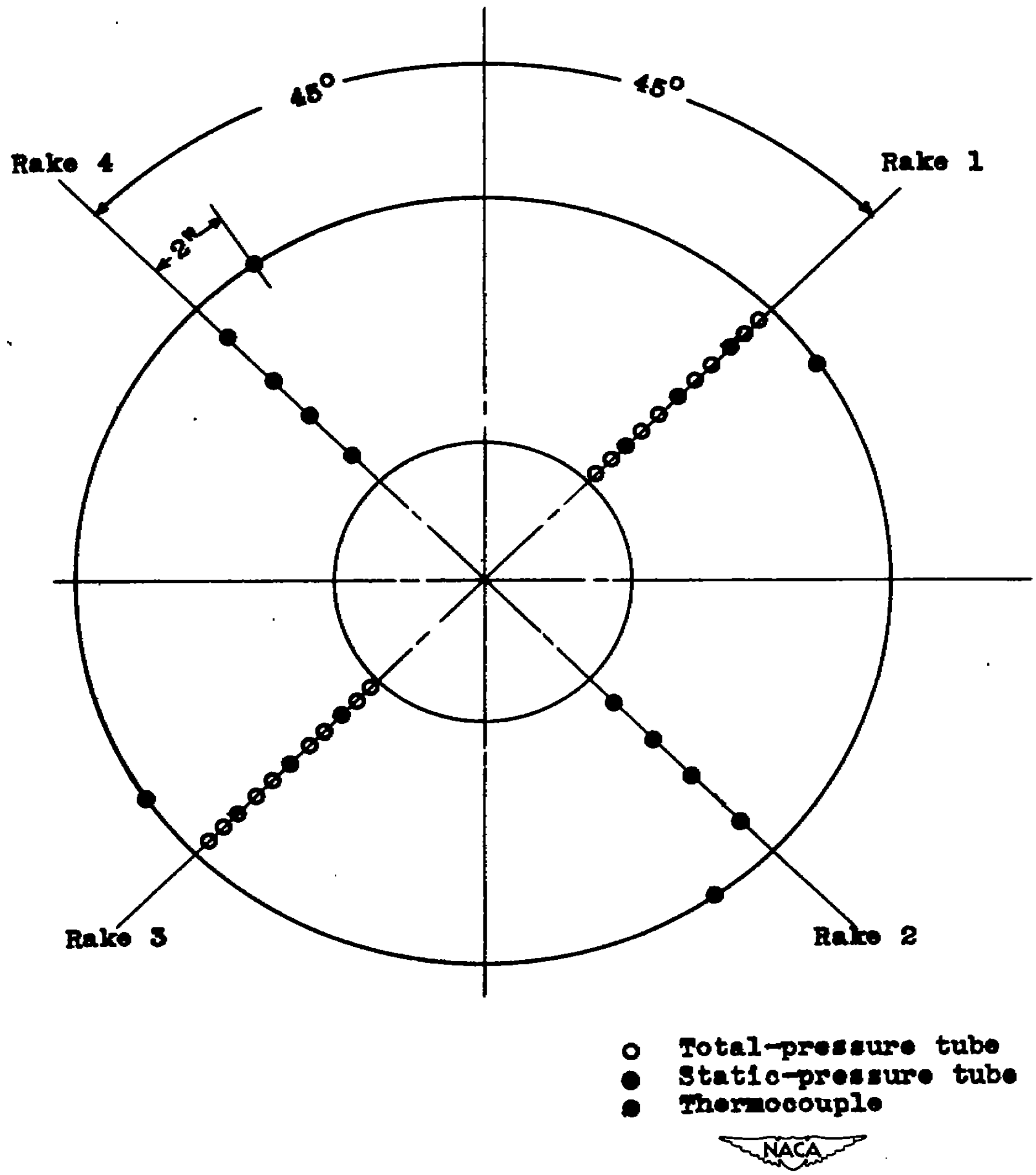


Figure 3. - Location of instrumentation at compressor inlet, station 2, $\frac{3}{4}$ inch behind rear flange of oil cooler.

[redacted]

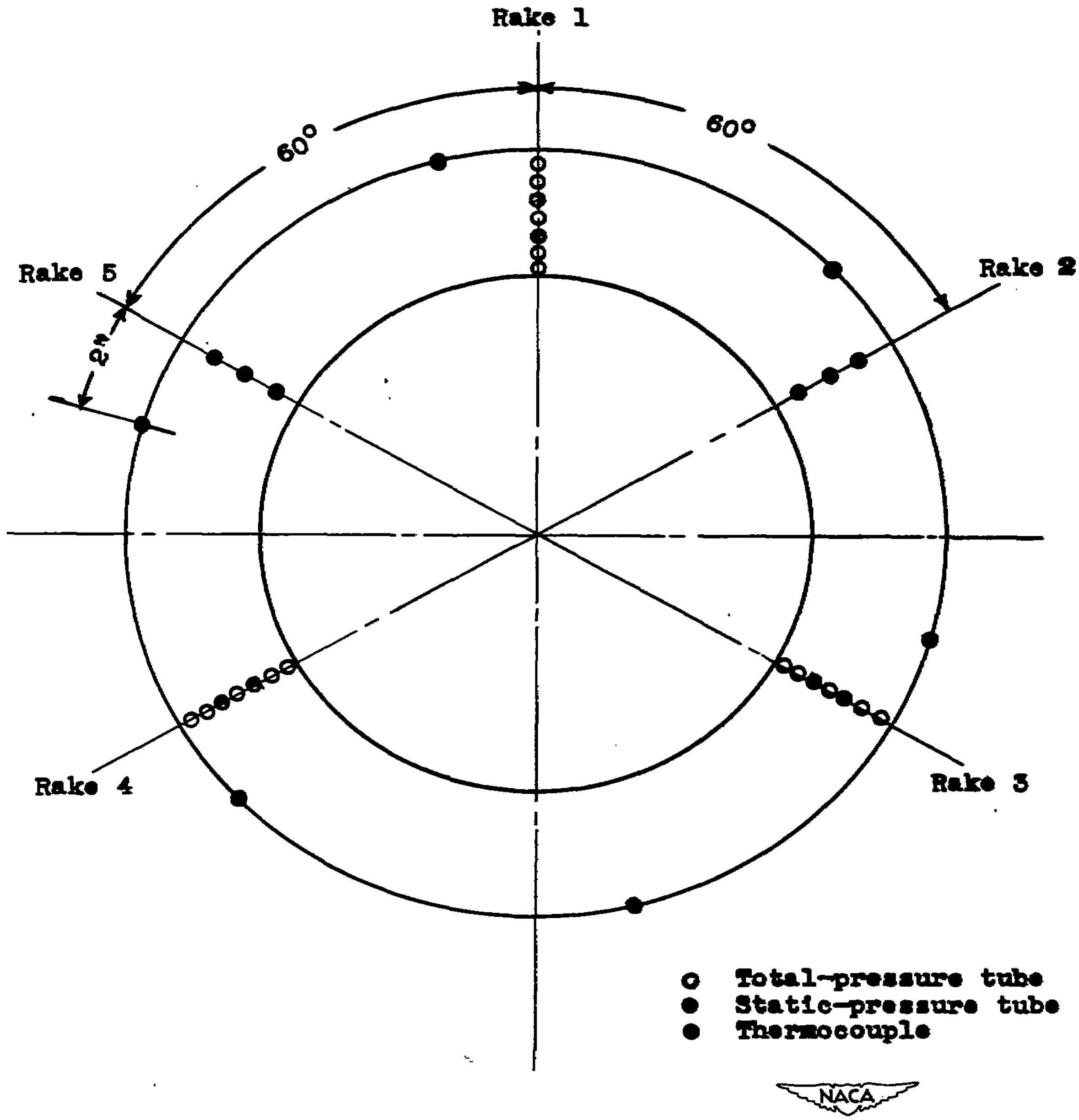
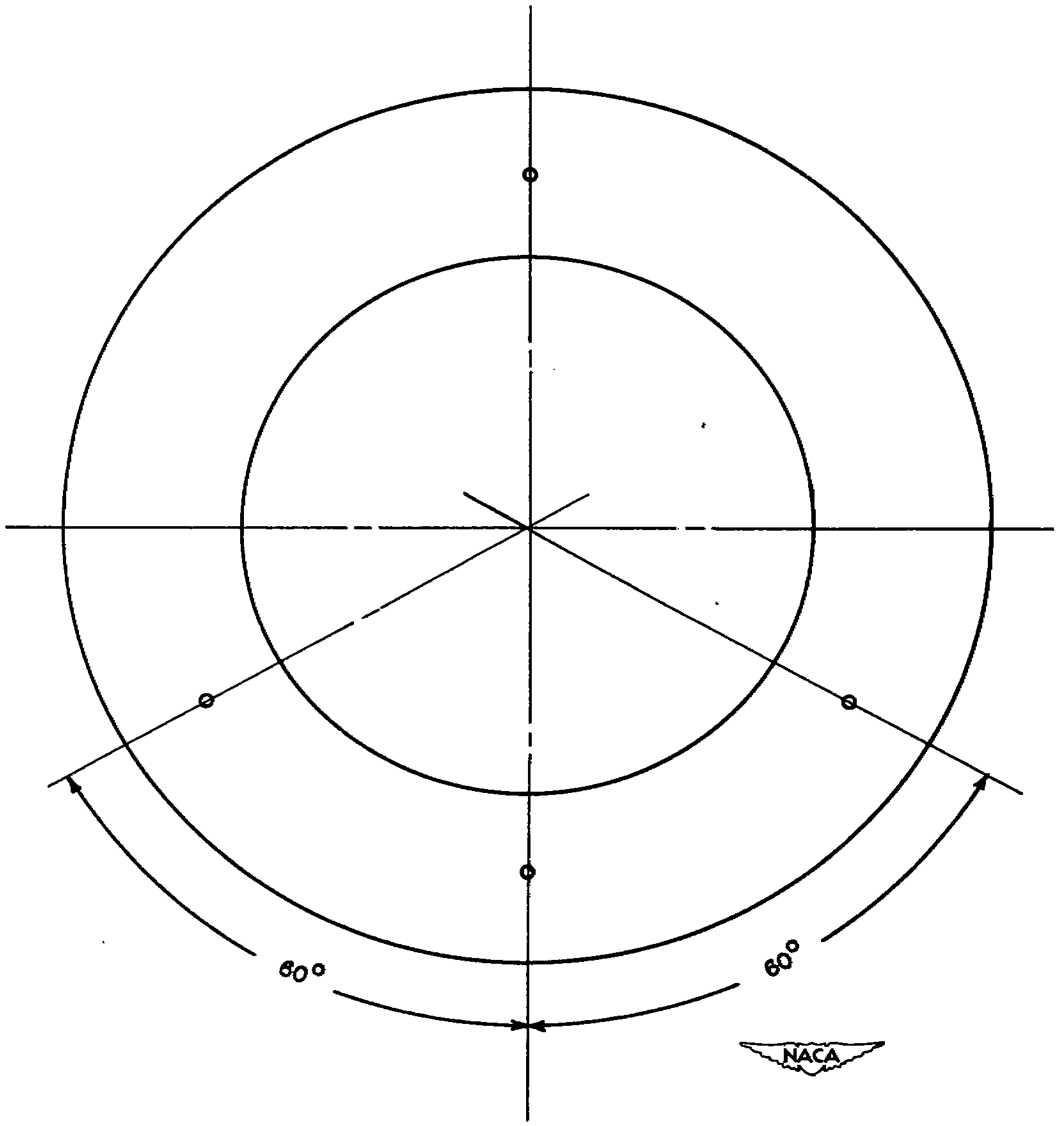
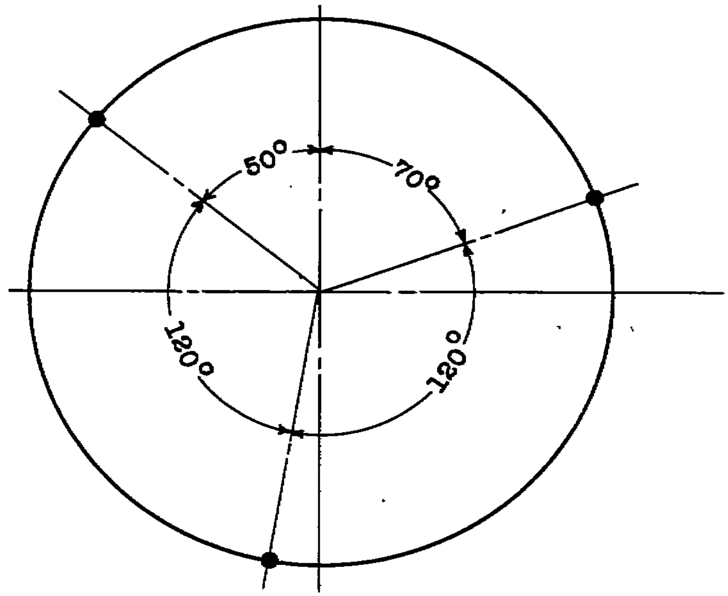


Figure 4. - Location of instrumentation at compressor outlet, station 4, $1\frac{1}{2}$ inches behind trailing edge of compressor-outlet straightening vanes.

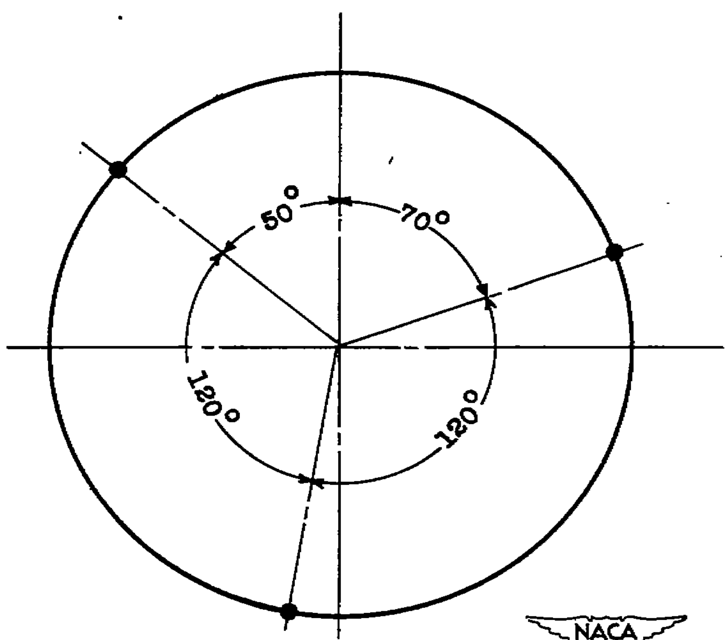


o Total-pressure tube

Figure 5. - Location of instrumentation at turbine inlet, station 5, $2\frac{3}{4}$ inches in front of center line of first-stage turbine stator blade.



First stage



Second stage

Figure 6. - Location of static-pressure tubes at turbine stator stages, station 6, center line of each stator blade.

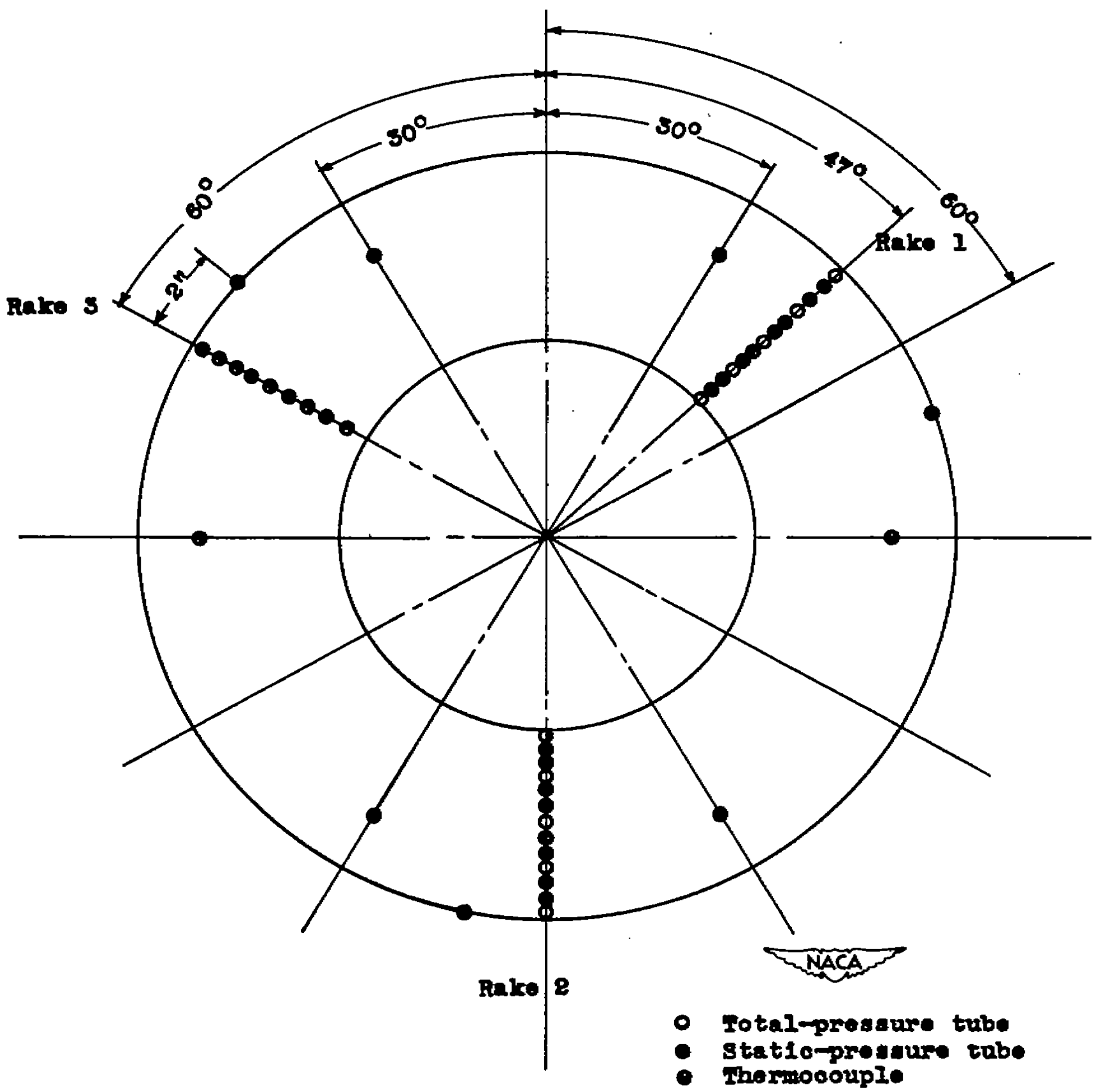
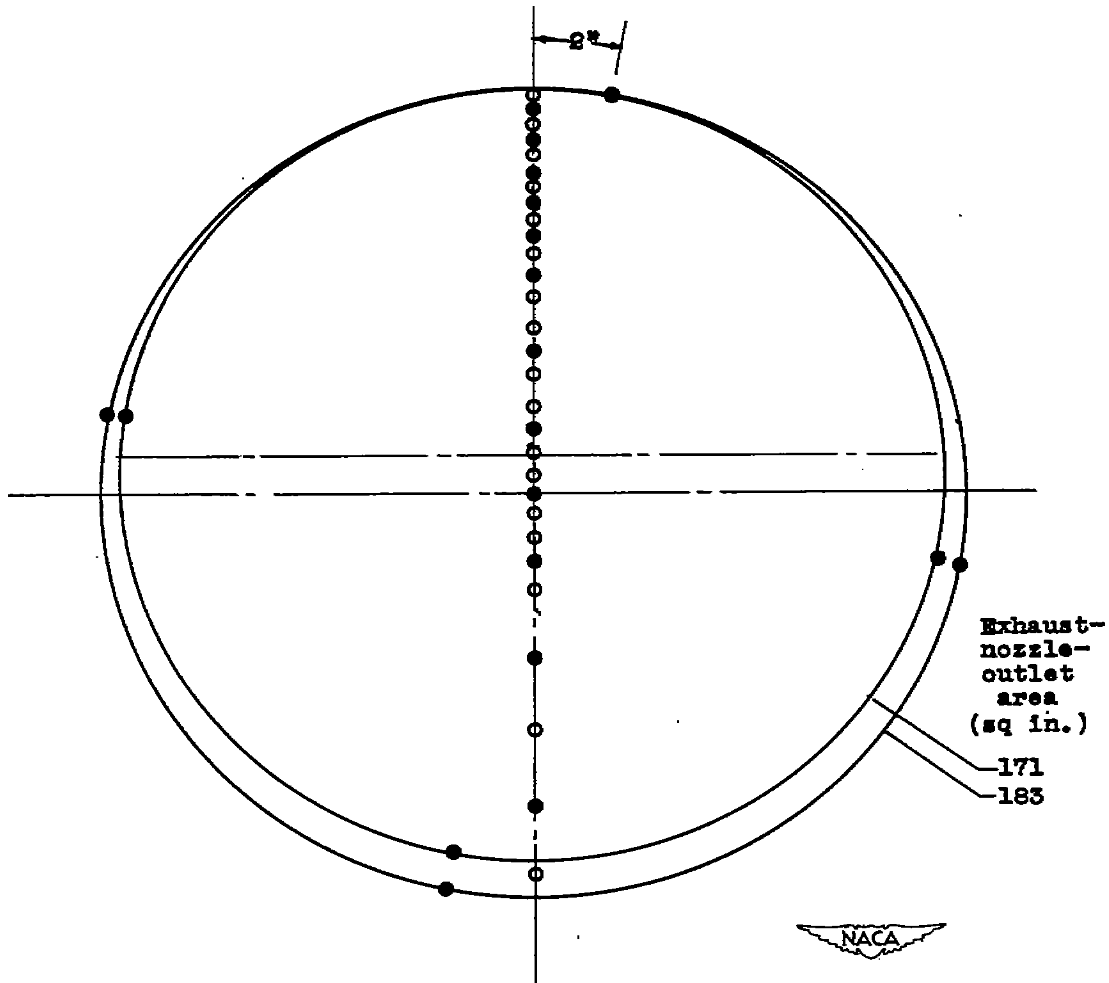


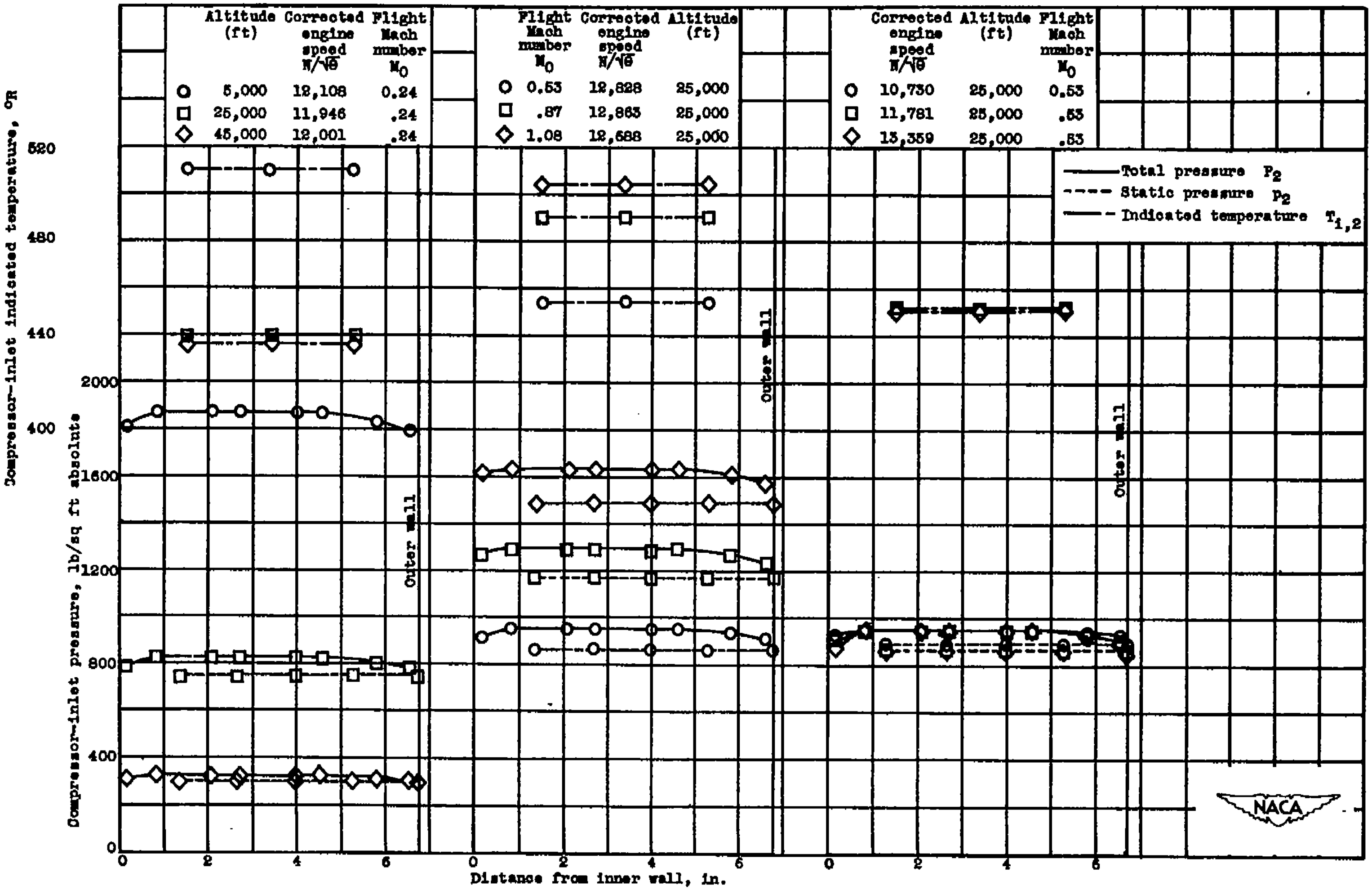
Figure 7. - Location of instrumentation at turbine outlet, station 7, $3\frac{3}{4}$ inches behind rear flange of turbine casing.

944



o Total-pressure tube
● Static-pressure tube
● Thermocouple

Figure 8. - Location of instrumentation in exhaust-nozzle outlet, station 8, 1 inch in front of rear edge of exhaust-nozzle outlet.

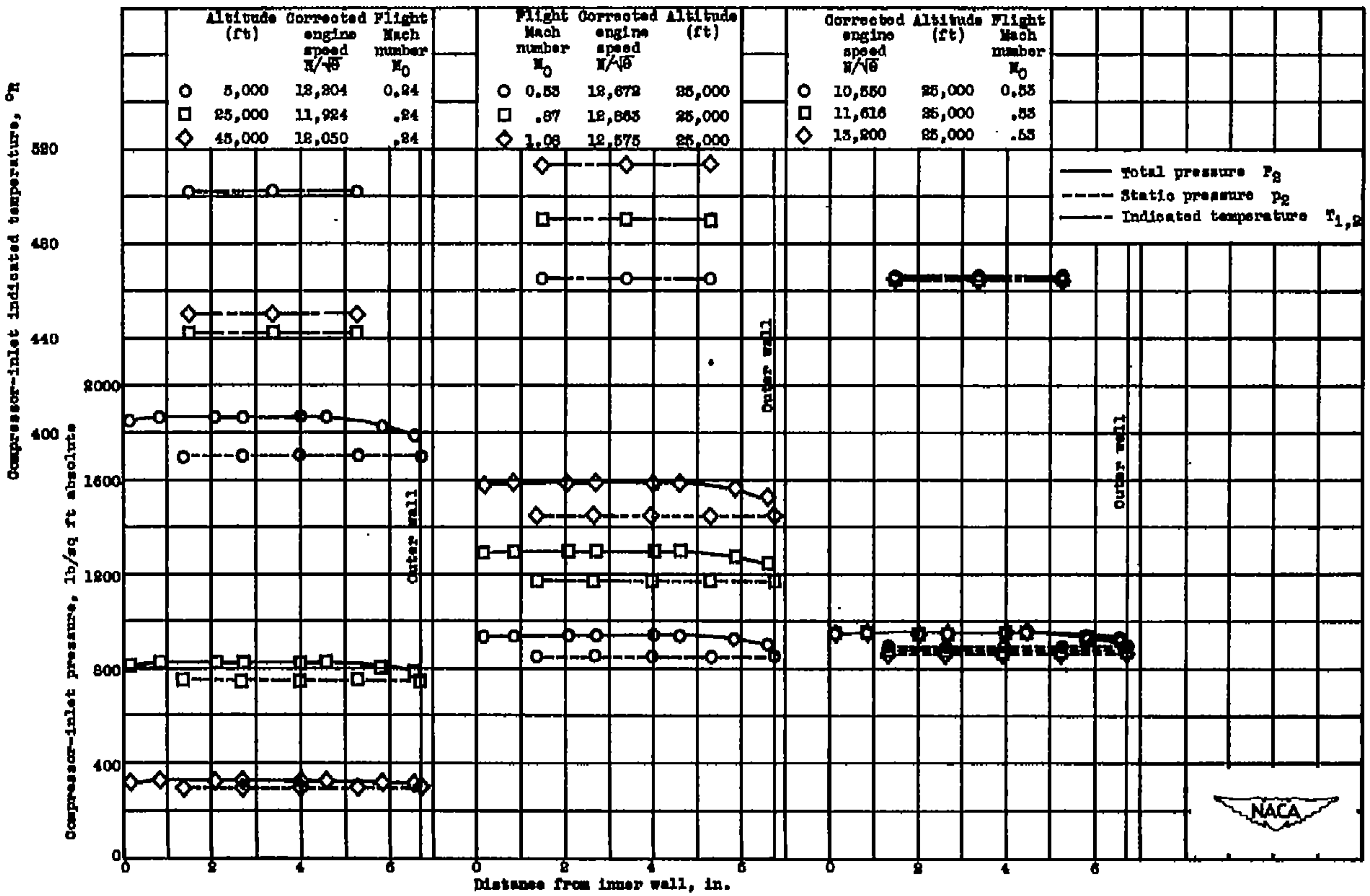


(a) Original engine.

Figure 9. - Effect of altitude, flight Mach number, and engine speed on radial distribution of total pressure, static pressure, and indicated temperature at compressor inlet, station 2.

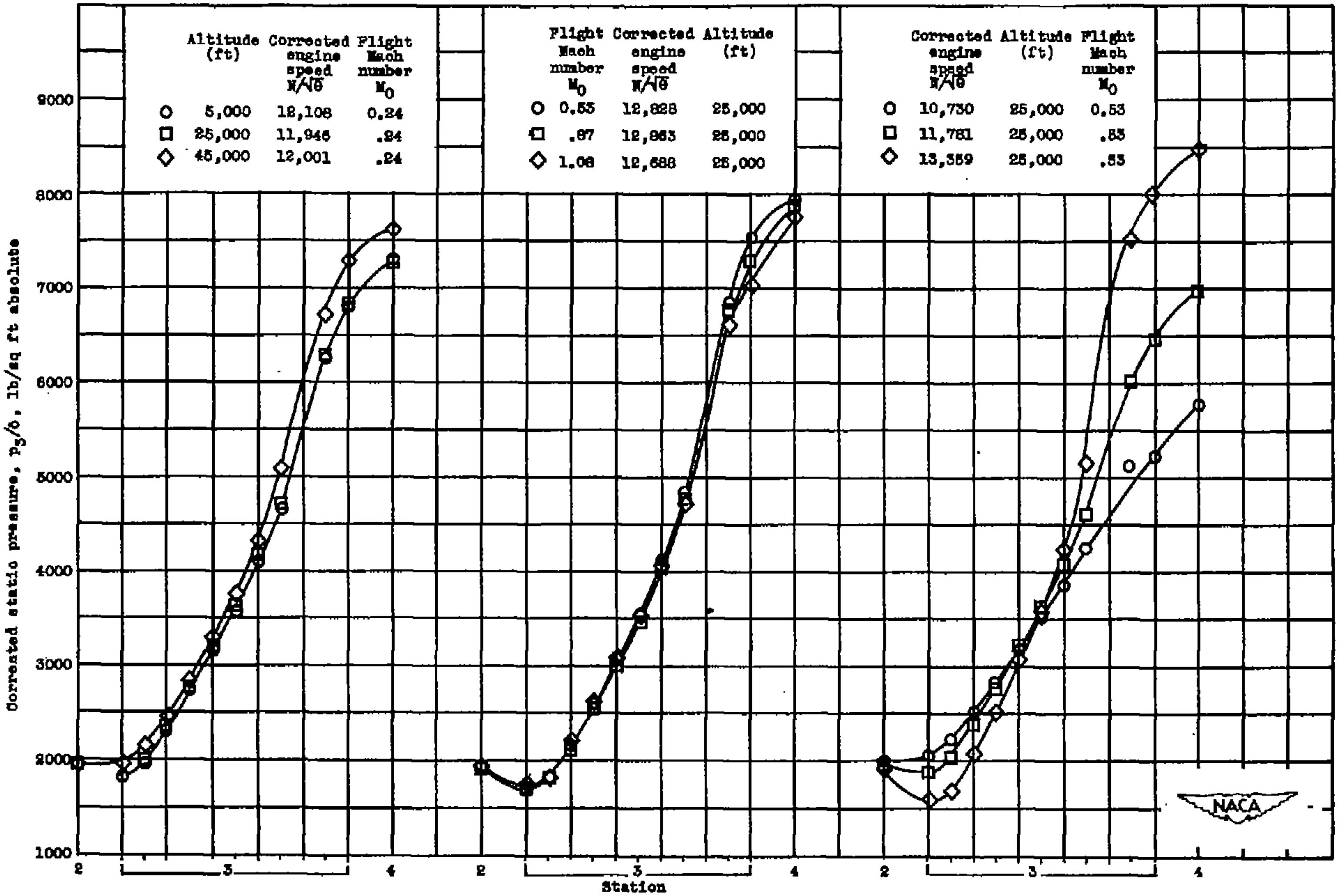
[redacted]

[redacted]



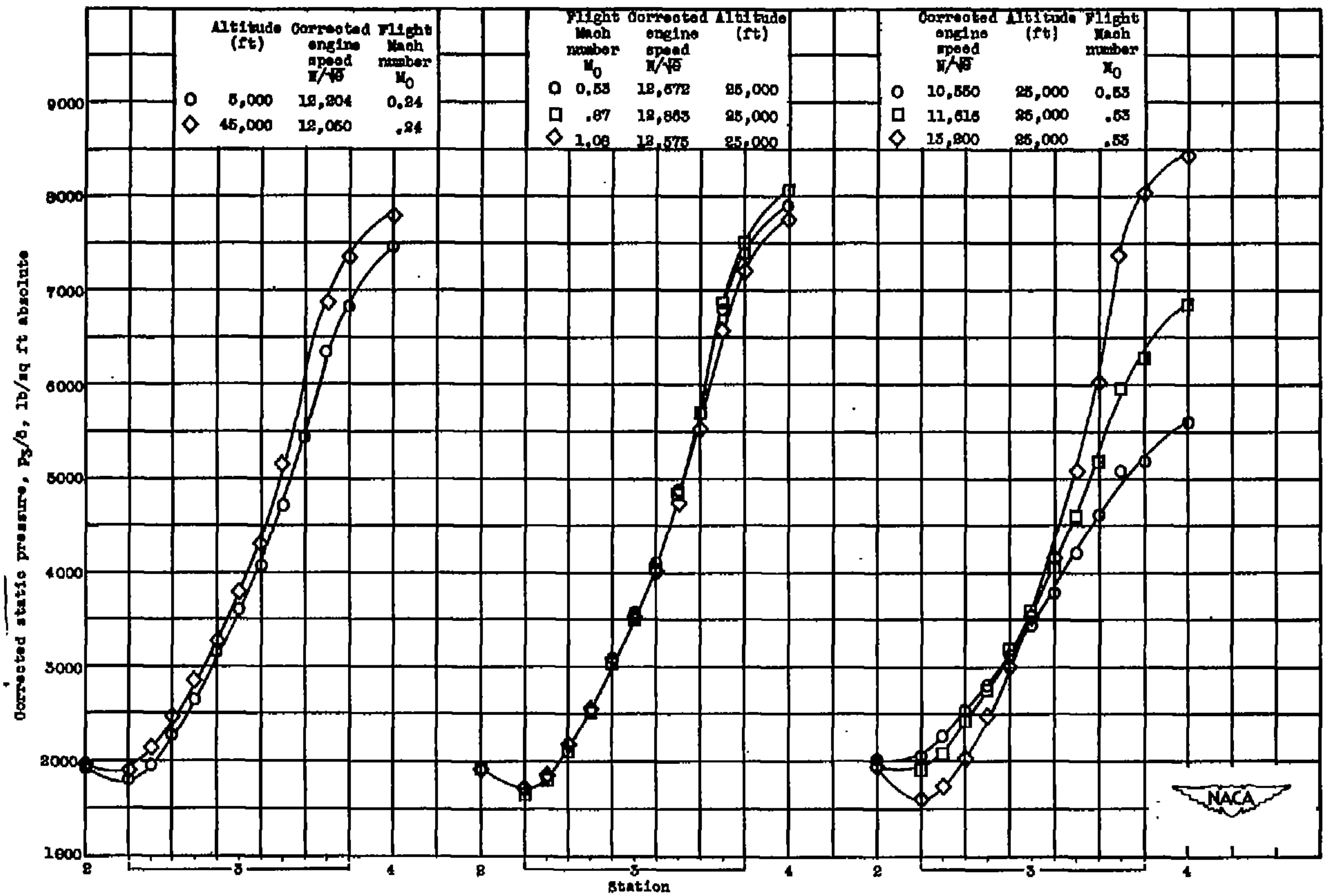
(b) Modified engine.

Figure 9. - Concluded. Effect of altitude, flight Mach number, and engine speed on radial distribution of total pressure, static pressure, and indicated temperature at compressor inlet, station 2.



(a) Original engine.

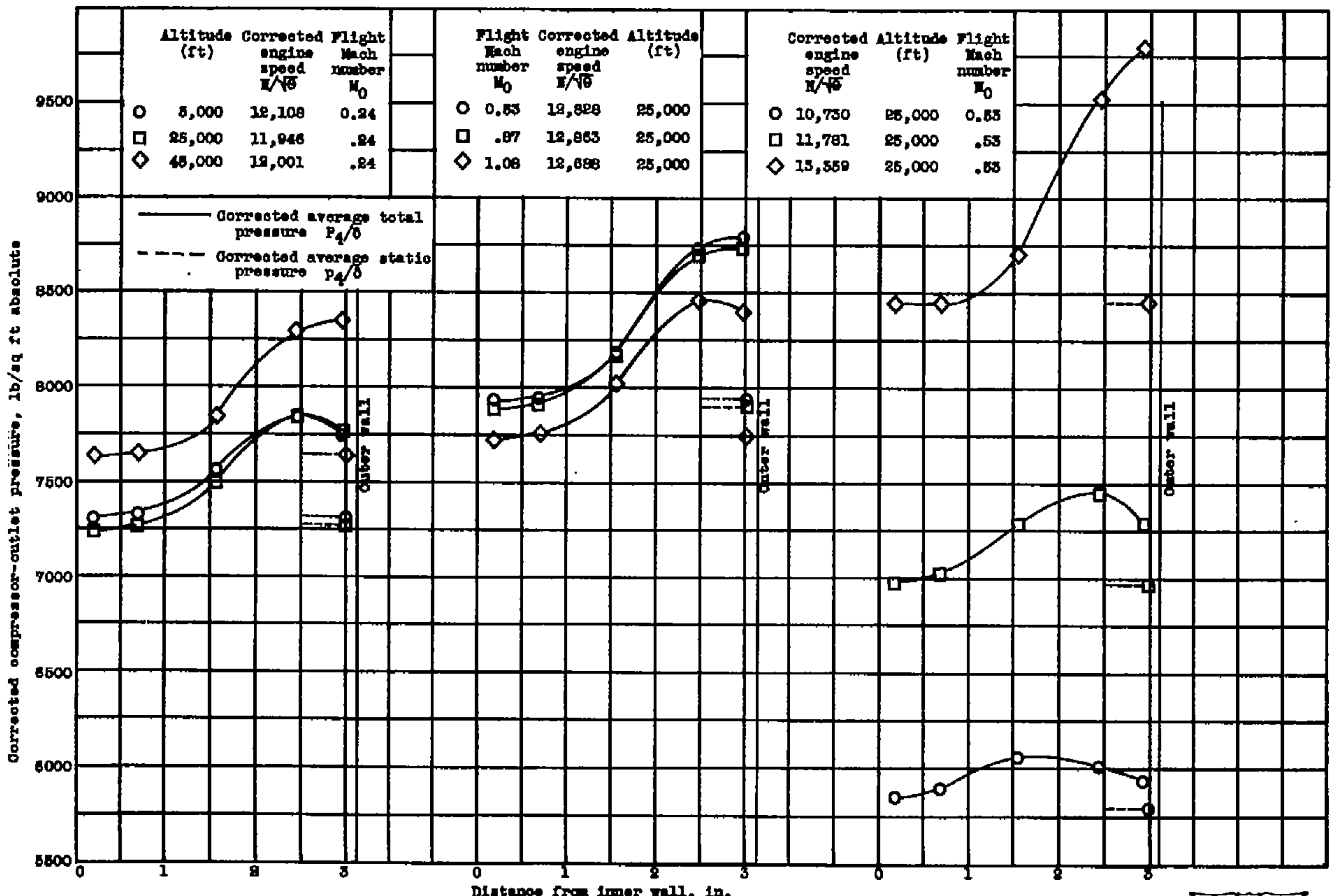
Figure 10. - Effect of altitude, flight Mach number, and engine speed on corrected static pressure at compressor stator stages, station 3.



(b) Modified engine.

Figure 10. - Concluded. Effect of altitude, flight Mach number, and engine speed on corrected static pressure at compressor stator stages, station 3.

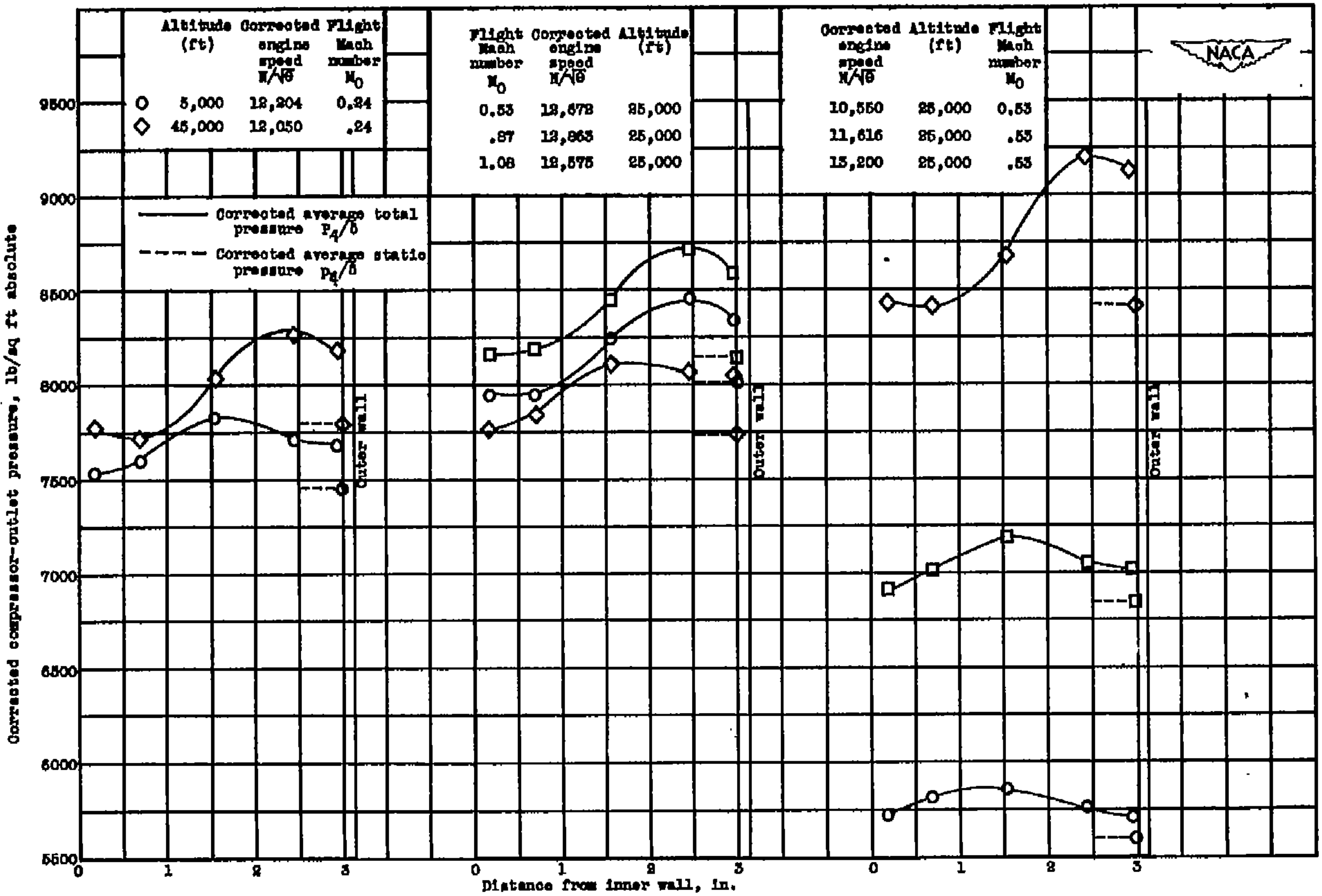
[redacted]



(a) Original engine.

Figure 11. - Effect of altitude, flight Mach number, and engine speed on radial distribution of average corrected total pressure at compressor outlet, station 4.

[redacted]

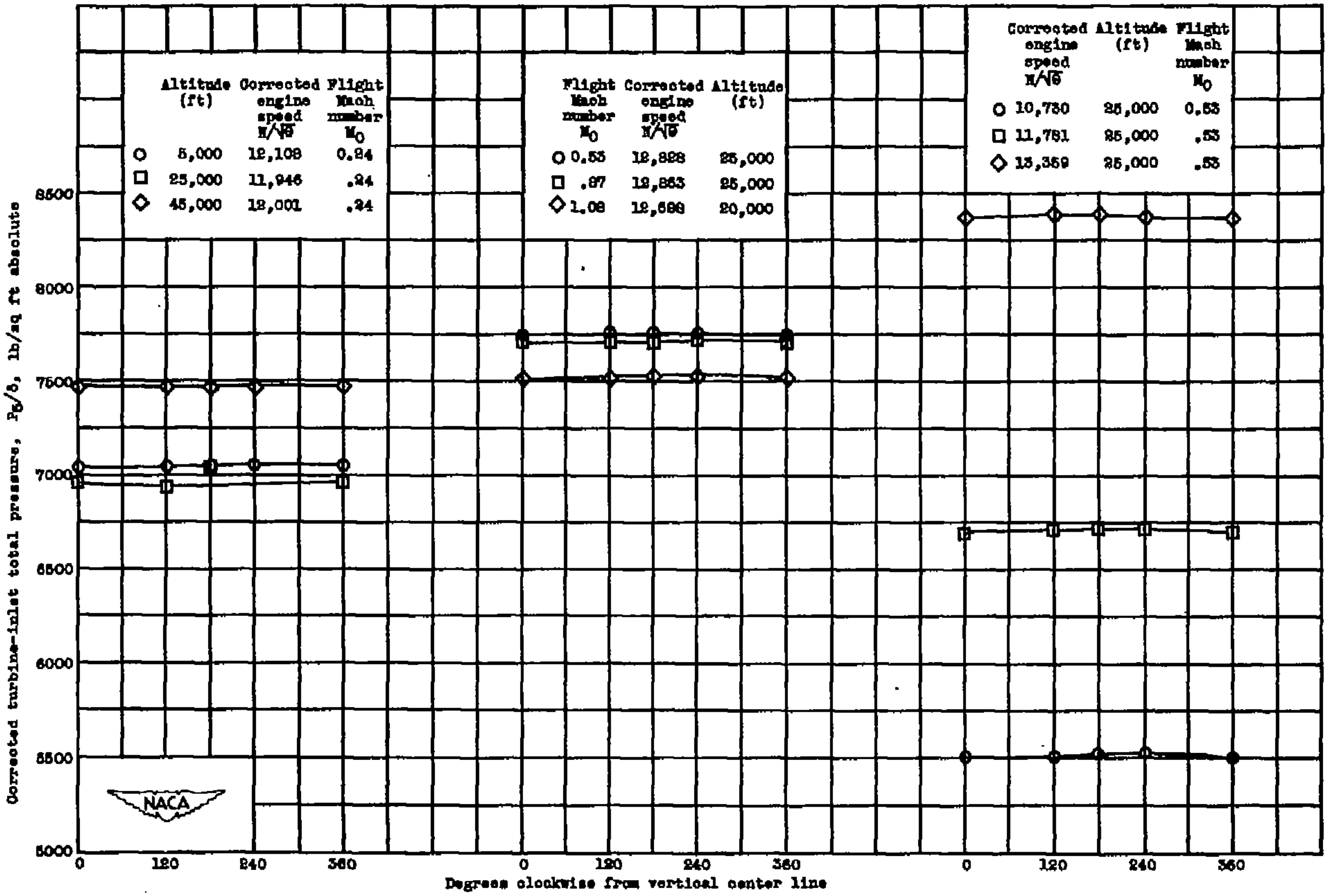


(b) Modified engine.

Figure 11. - Concluded. Effect of altitude, flight Mach number, and engine speed on radial distribution of average corrected total pressure at compressor outlet, station 4.

CONFIDENTIAL

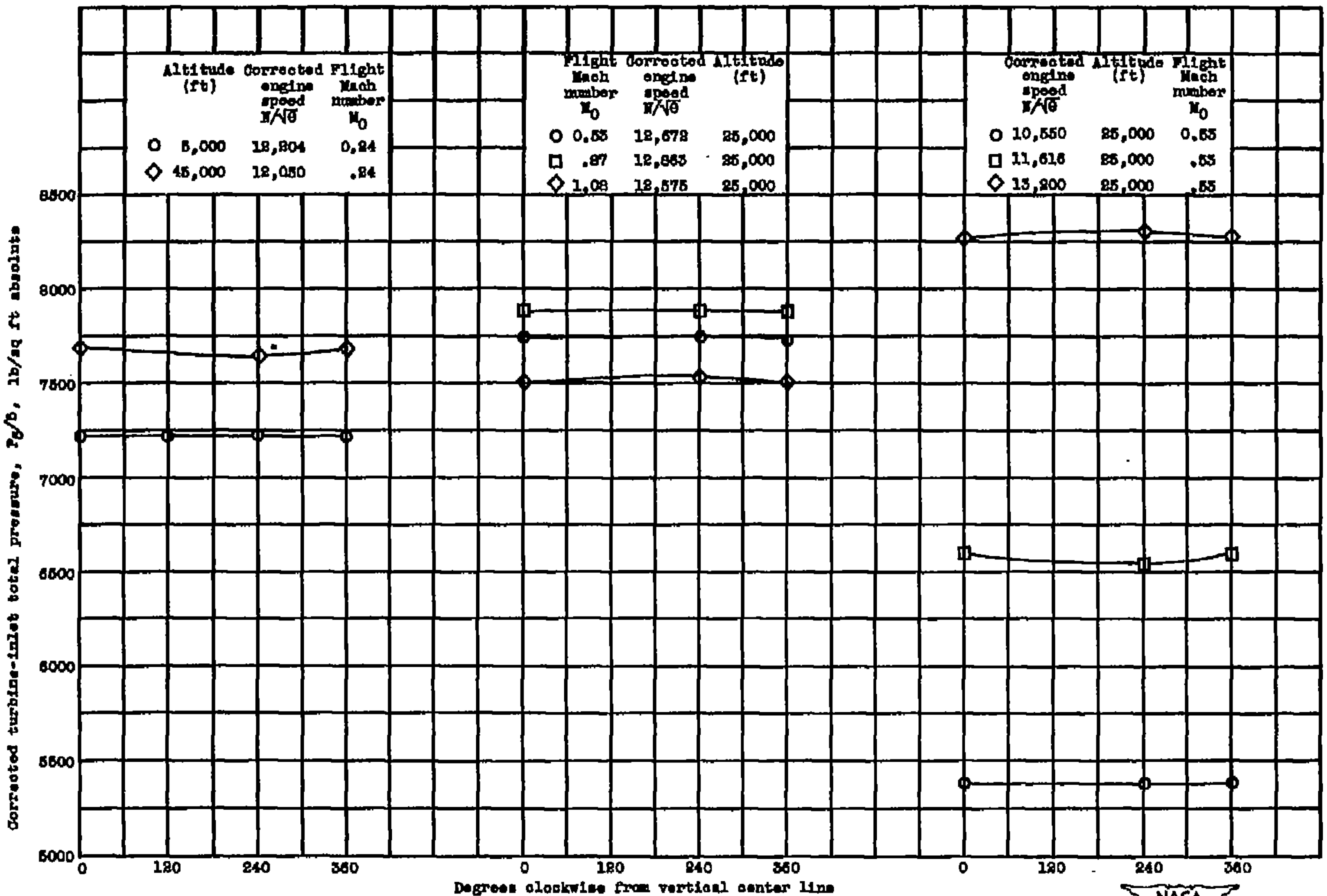
[redacted]



(a) Original engine.

Figure 12. - Effect of altitude, flight Mach number, and engine speed on circumferential distribution of corrected total pressure at turbine inlet, station 5.

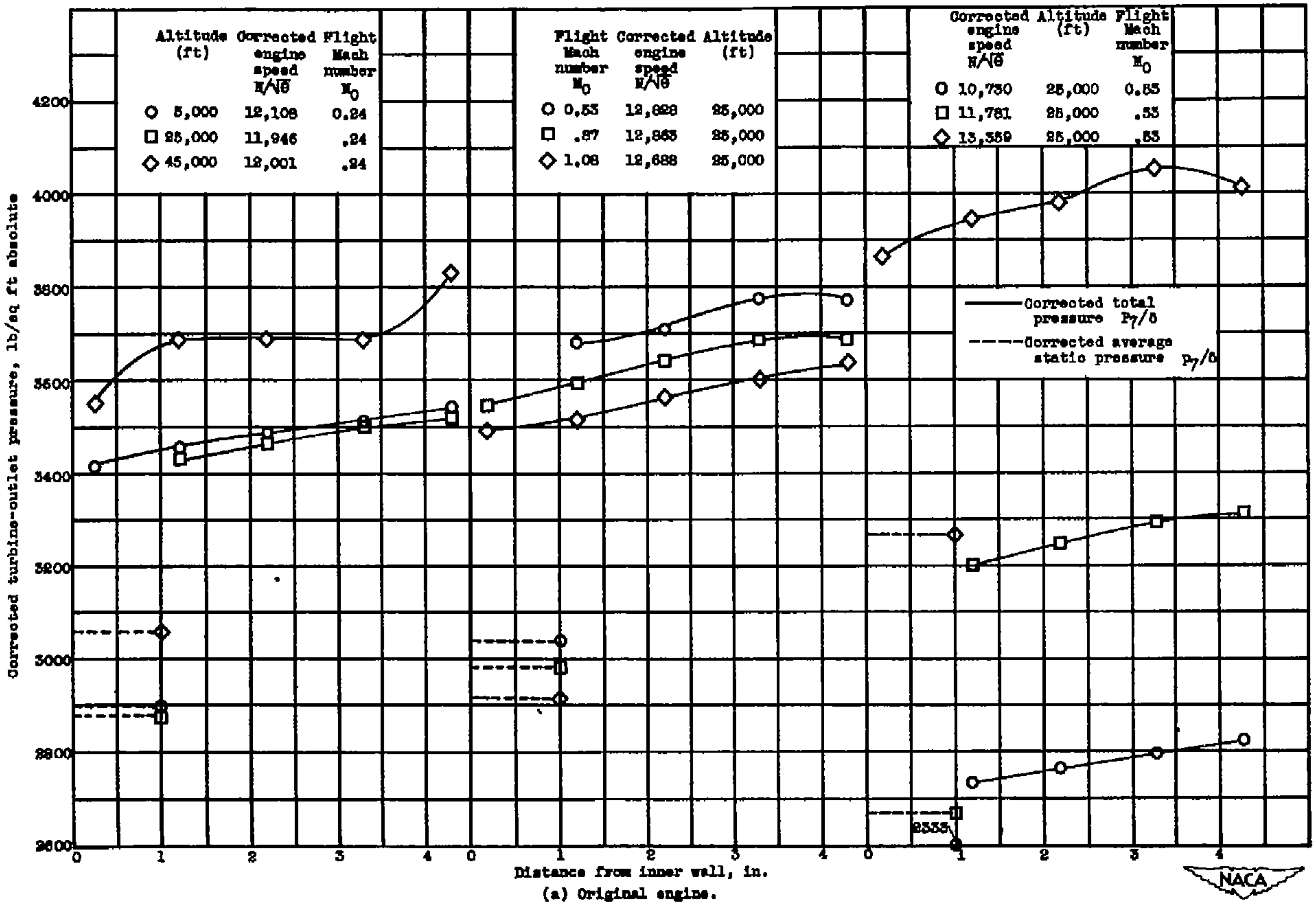
[redacted]



(b) Modified engine.

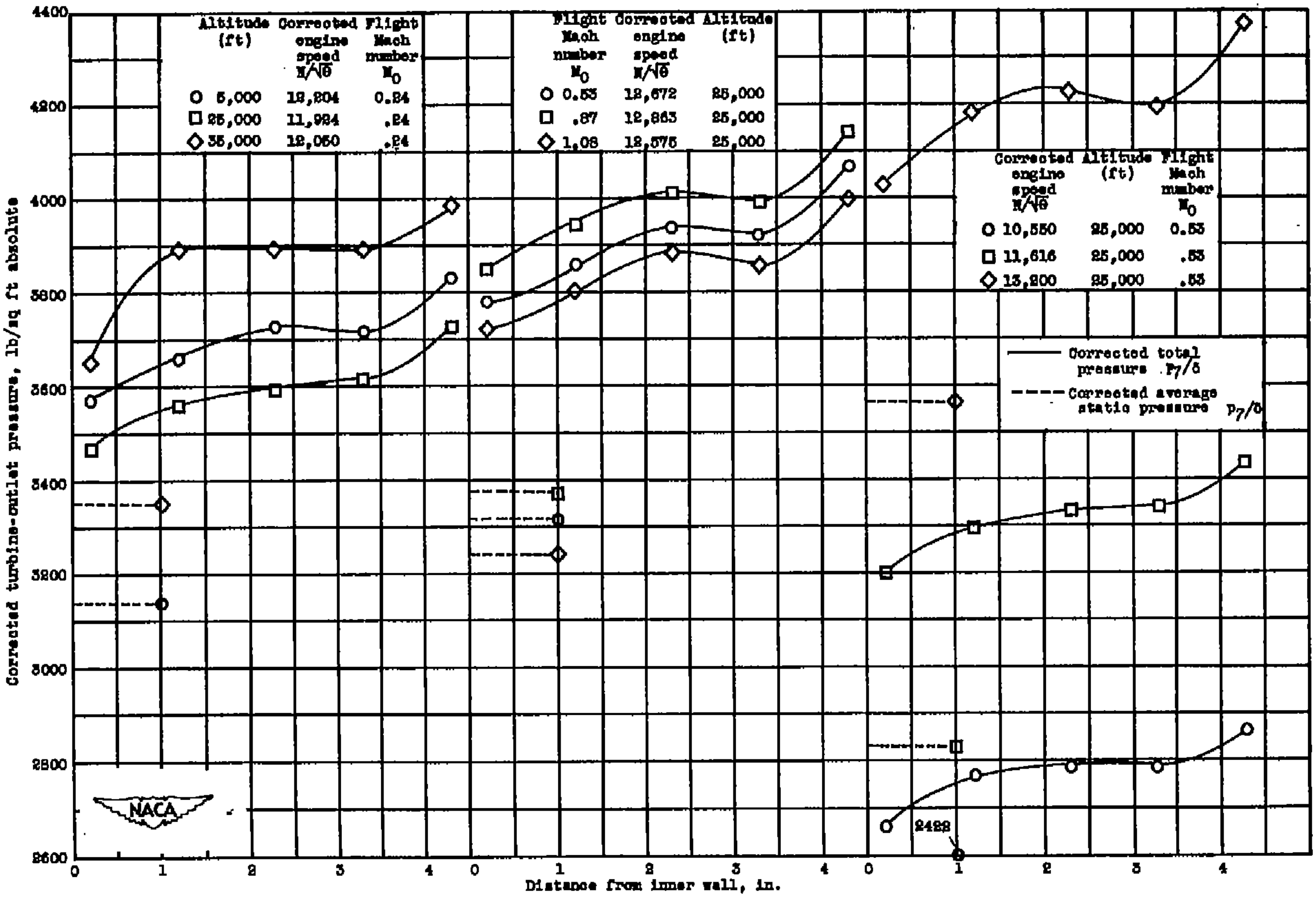
Figure 12. - Concluded. Effect of altitude, flight Mach number, and engine speed on circumferential distribution of corrected total pressure at turbine inlet, station 5.

[redacted]



(a) Original engine.

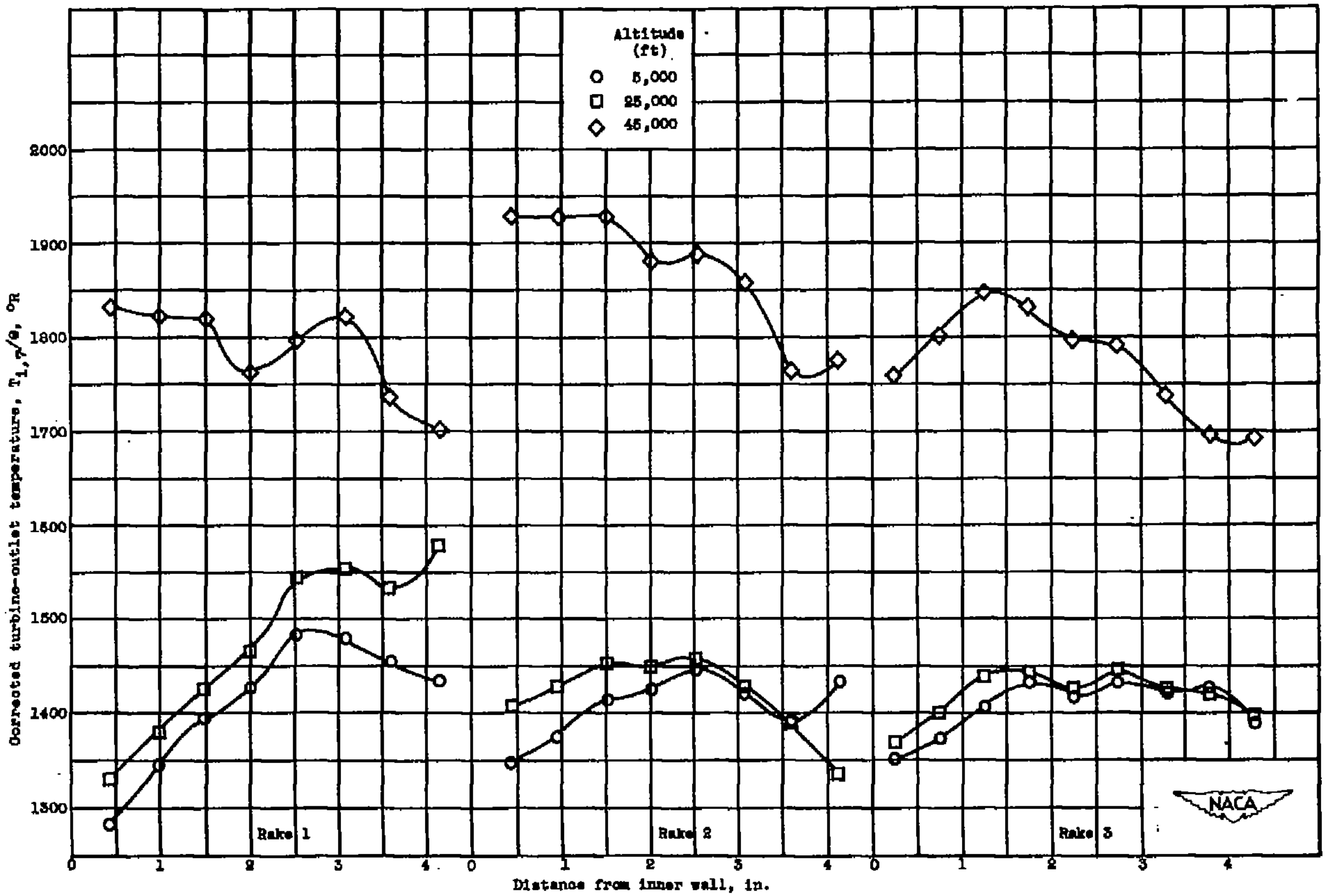
Figure 15. - Effect of altitude, flight Mach number, and engine speed on radial distribution of corrected total pressure at turbine outlet, station 7.



(b) Modified engine.

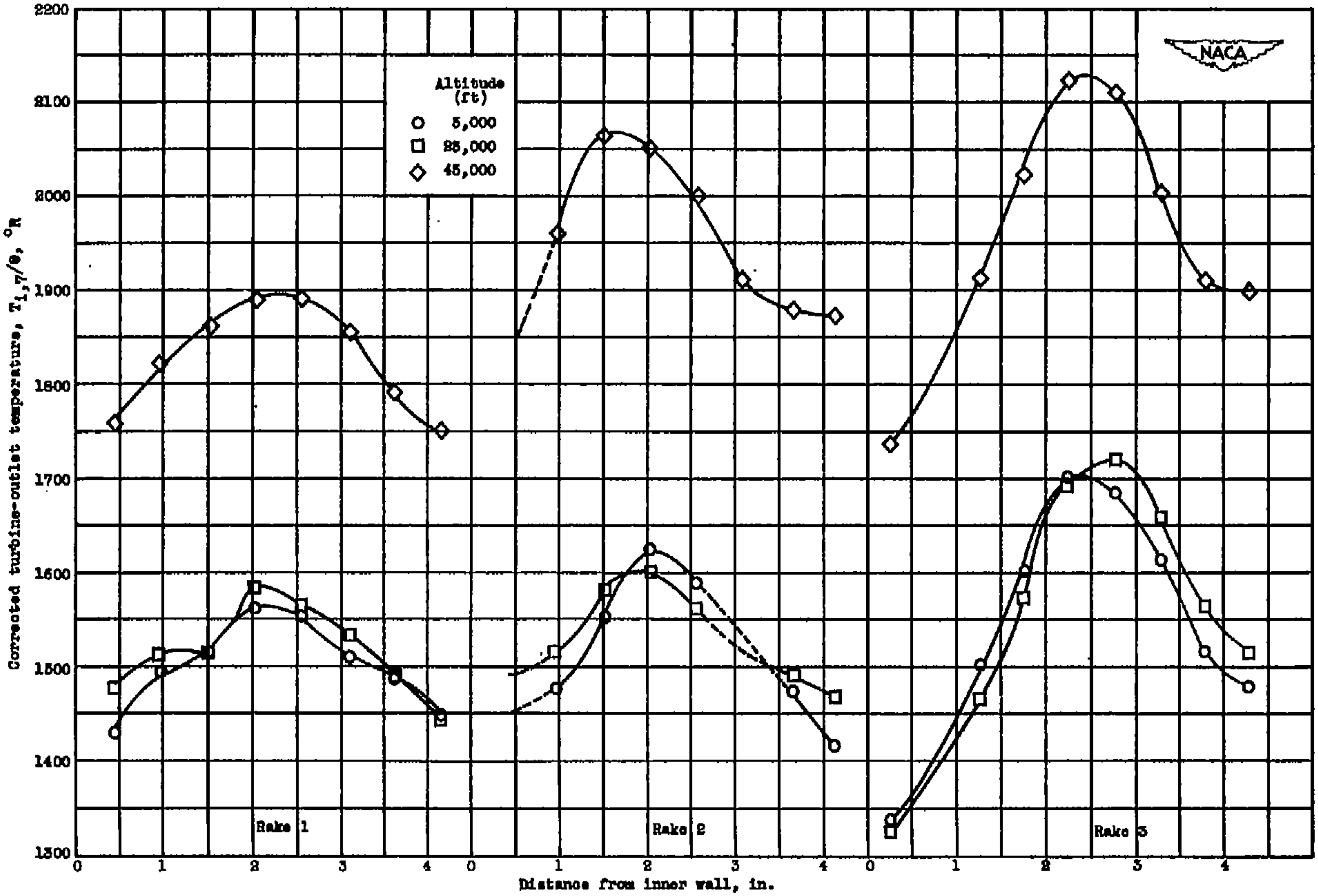
Figure 13. - Concluded. Effect of altitude, flight Mach number, and engine speed on radial distribution of corrected total pressure at turbine outlet, station 7.

[redacted]



(a) Original engine.

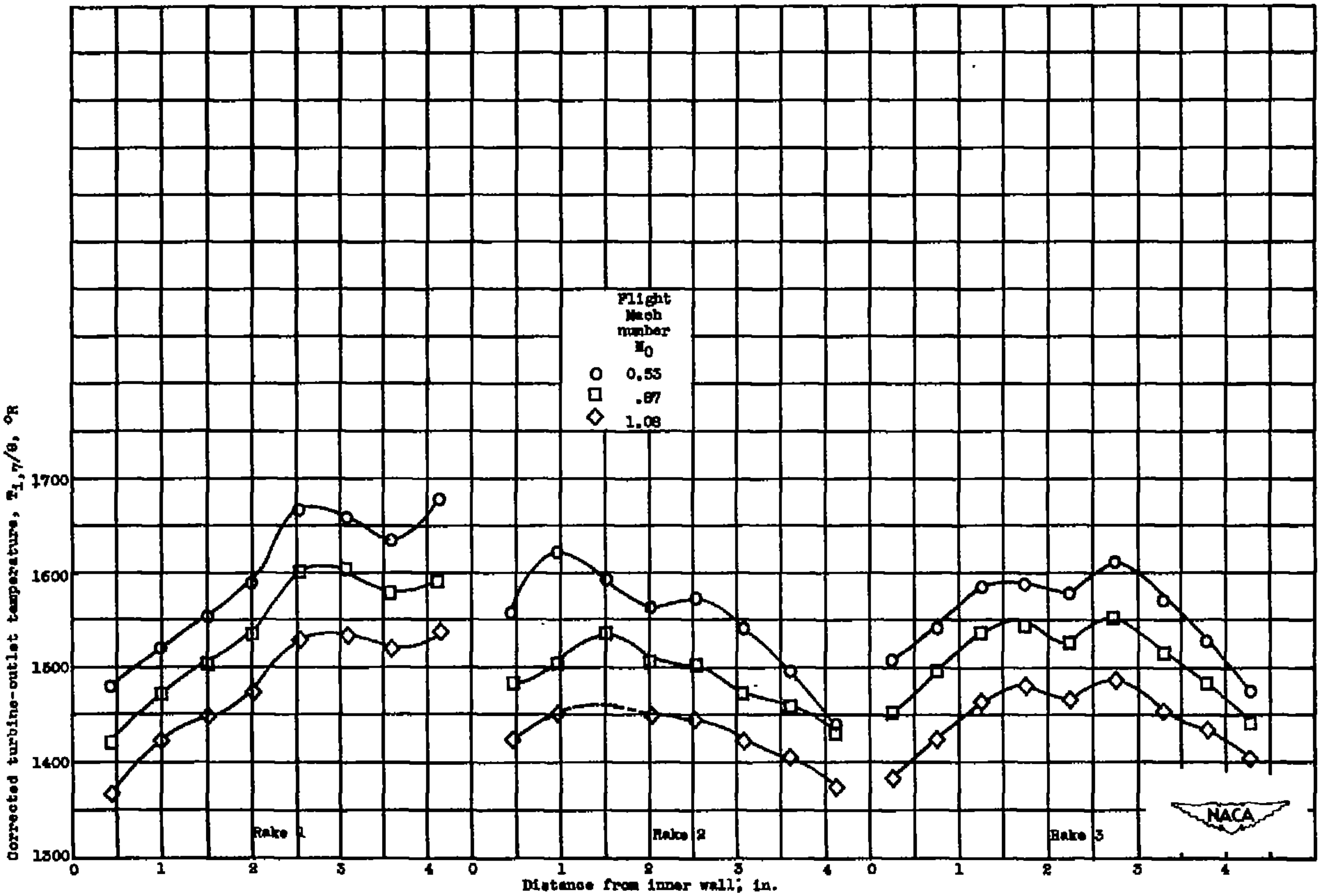
Figure 14. - Effect of altitude on radial distribution of corrected indicated temperature at turbine outlet, station 7. Flight Mach number, 0.24; corrected engine speed, approximately 12,000 rpm.



(b) Modified engine.

Figure 14. - Concluded. Effect of altitude on radial distribution of corrected indicated temperature at turbine outlet, station 7. Flight Mach number, 0.24; corrected engine speed, approximately 12,000 rpm.

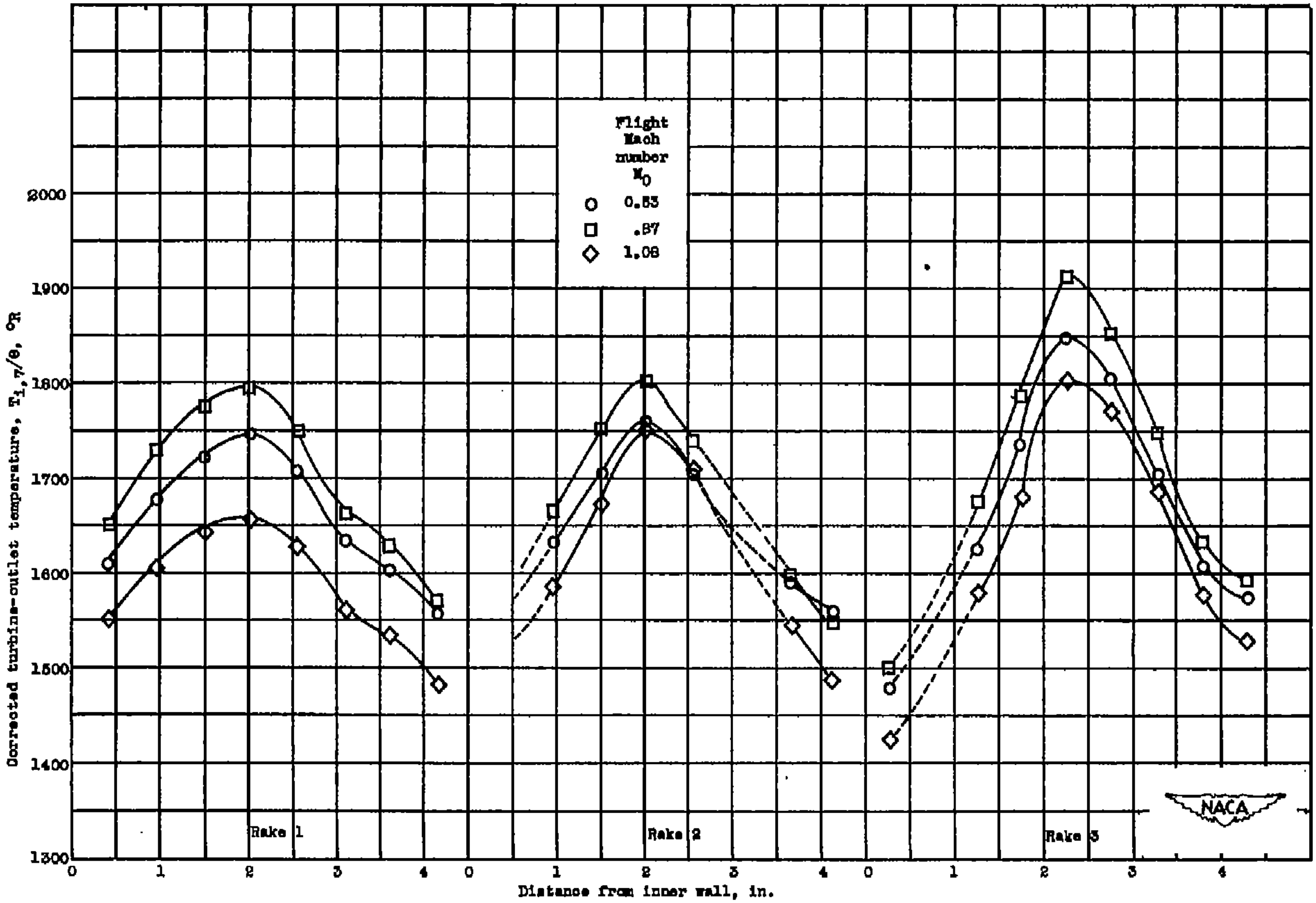
[redacted]



(a) Original engine.

Figure 15. - Effect of flight Mach number on radial distribution of corrected indicated temperature at turbine outlet, station 7. Altitude, 25,000 feet; corrected engine speed, approximately 12,700 rpm.

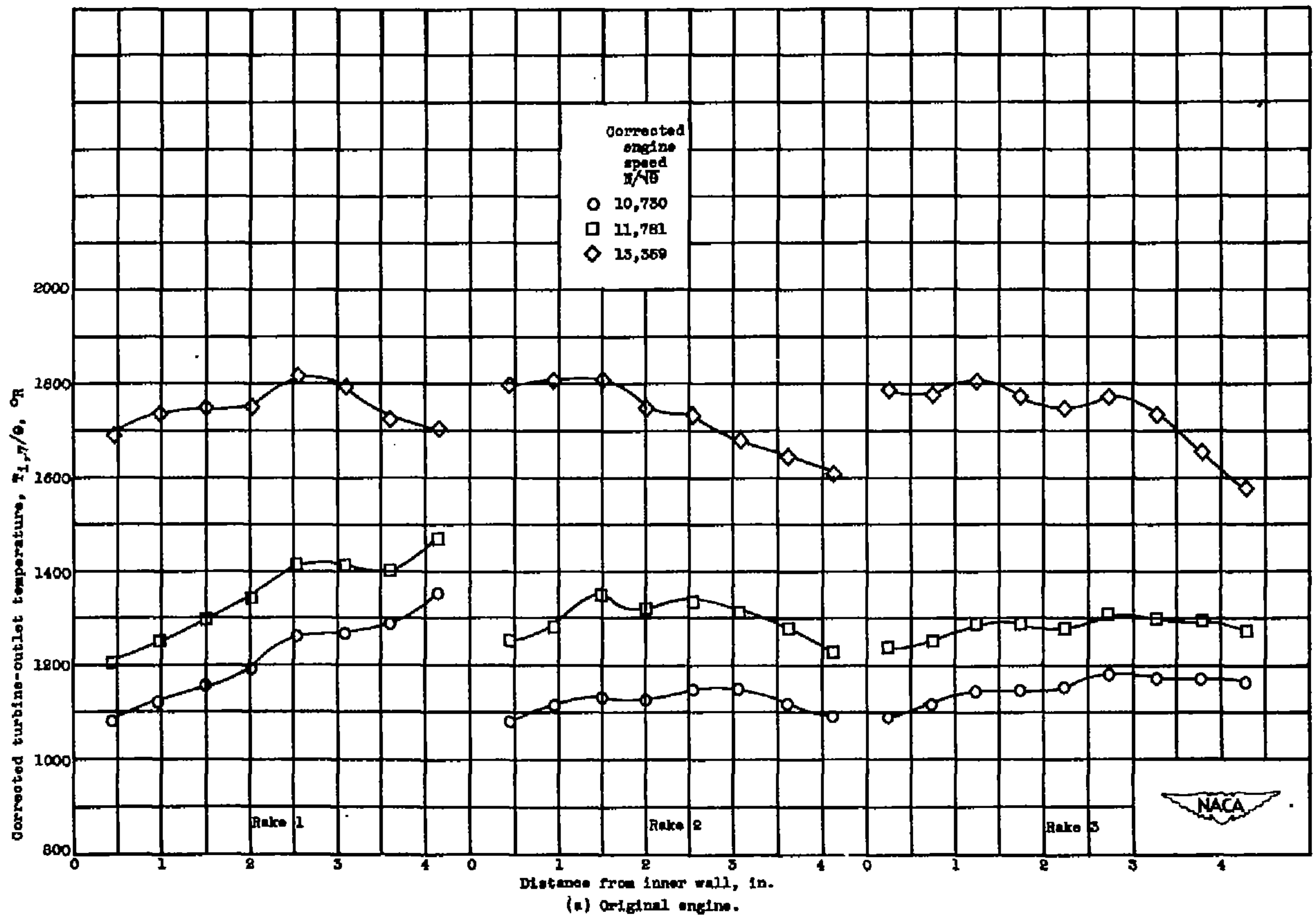
[redacted]



[redacted]

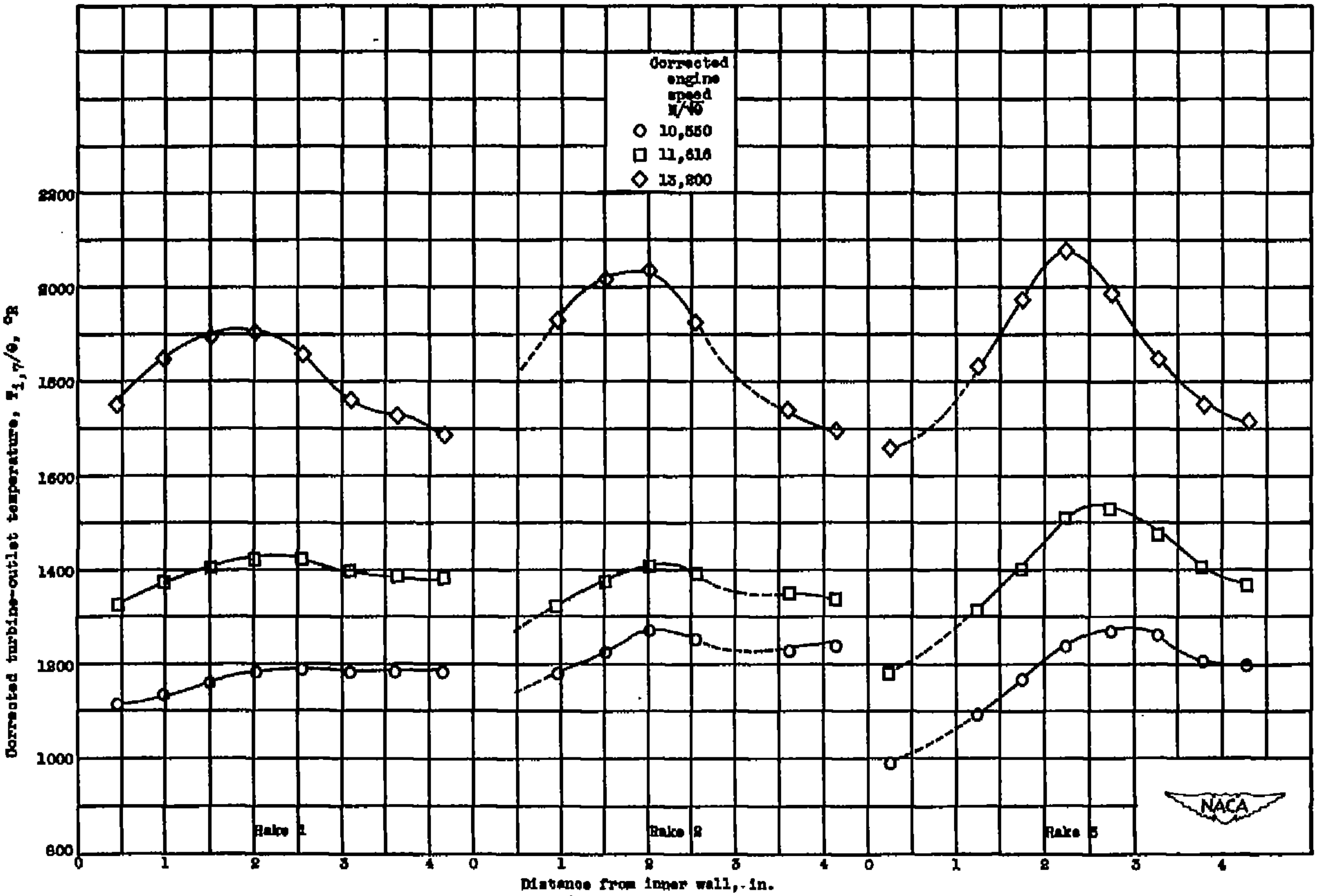
(b) Modified engine.

Figure 15. - Concluded. Effect of flight Mach number on radial distribution of corrected indicated temperature at turbine outlet, station 7. Altitude, 25,000 feet; corrected engine speed, approximately 12,700 rpm.



(a) Original engine.

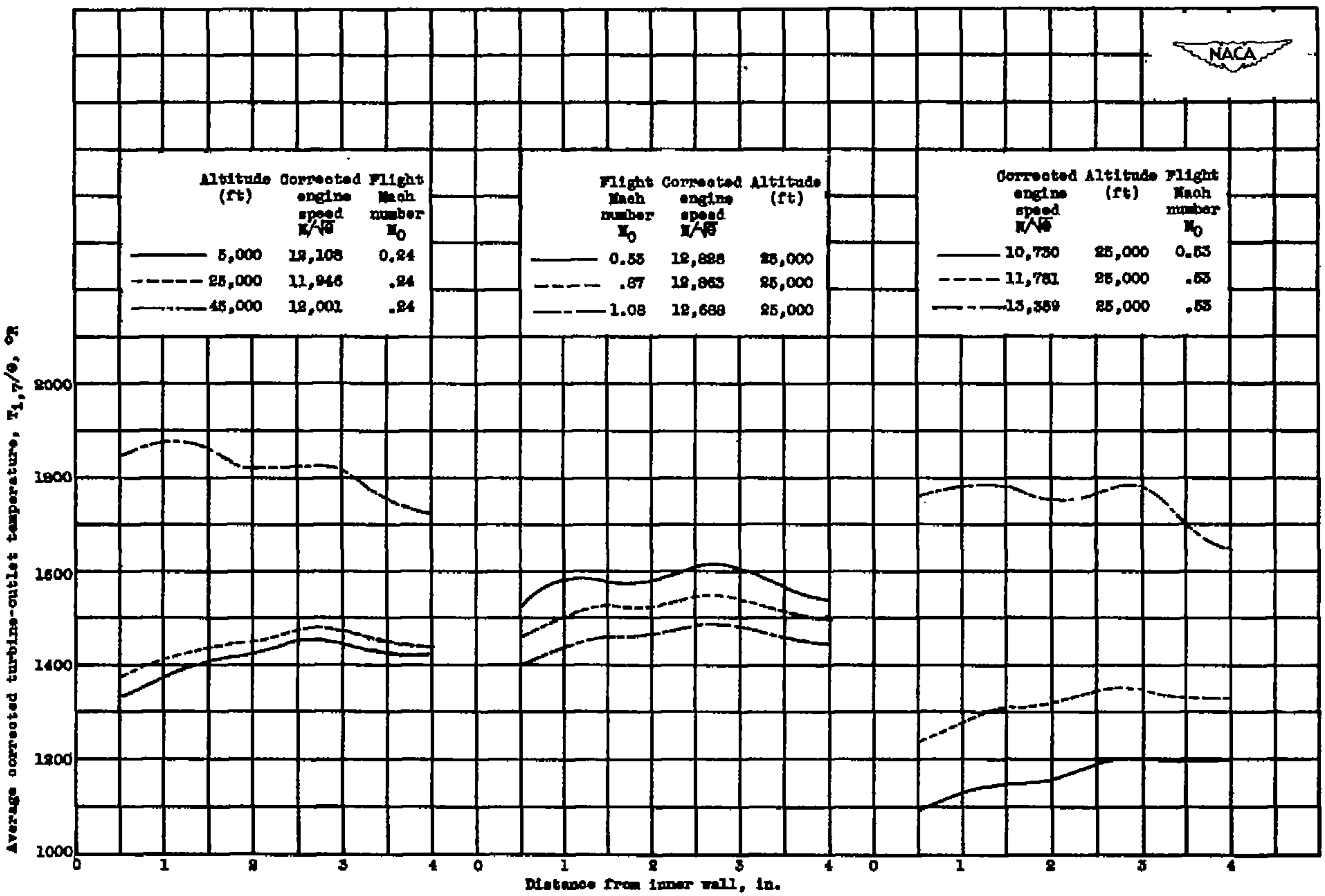
Figure 16. - Effect of engine speed on radial distribution of corrected indicated temperature at turbine outlet, station 7. Altitude, 25,000 feet; flight Mach number, 0.53.



(b) Modified engine.

Figure 16. - Concluded. Effect of engine speed on radial distribution of corrected indicated temperature at turbine outlet, station 7. Altitude, 25,000 feet; flight Mach number, 0.53.

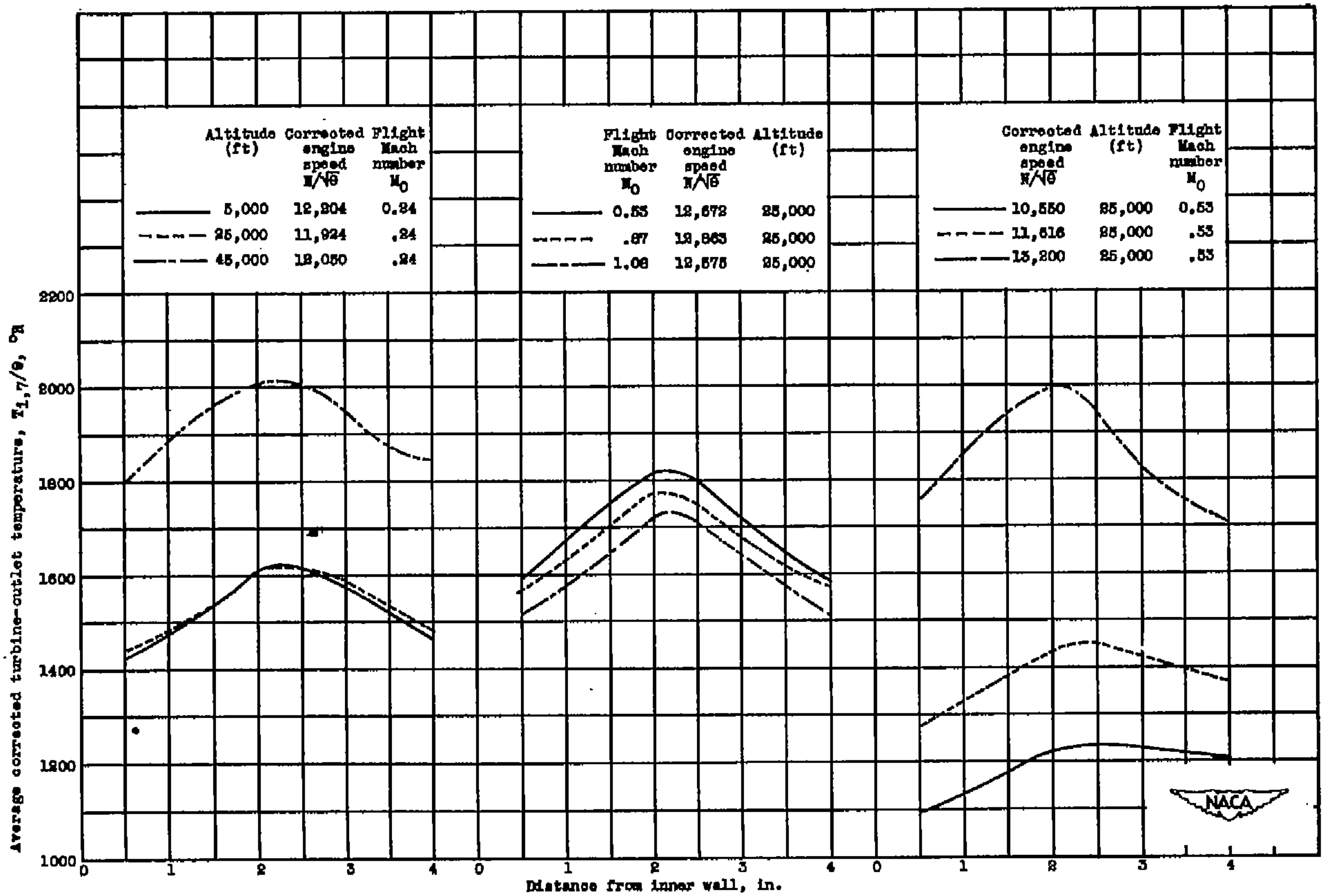
[redacted]



(a) Original engine.

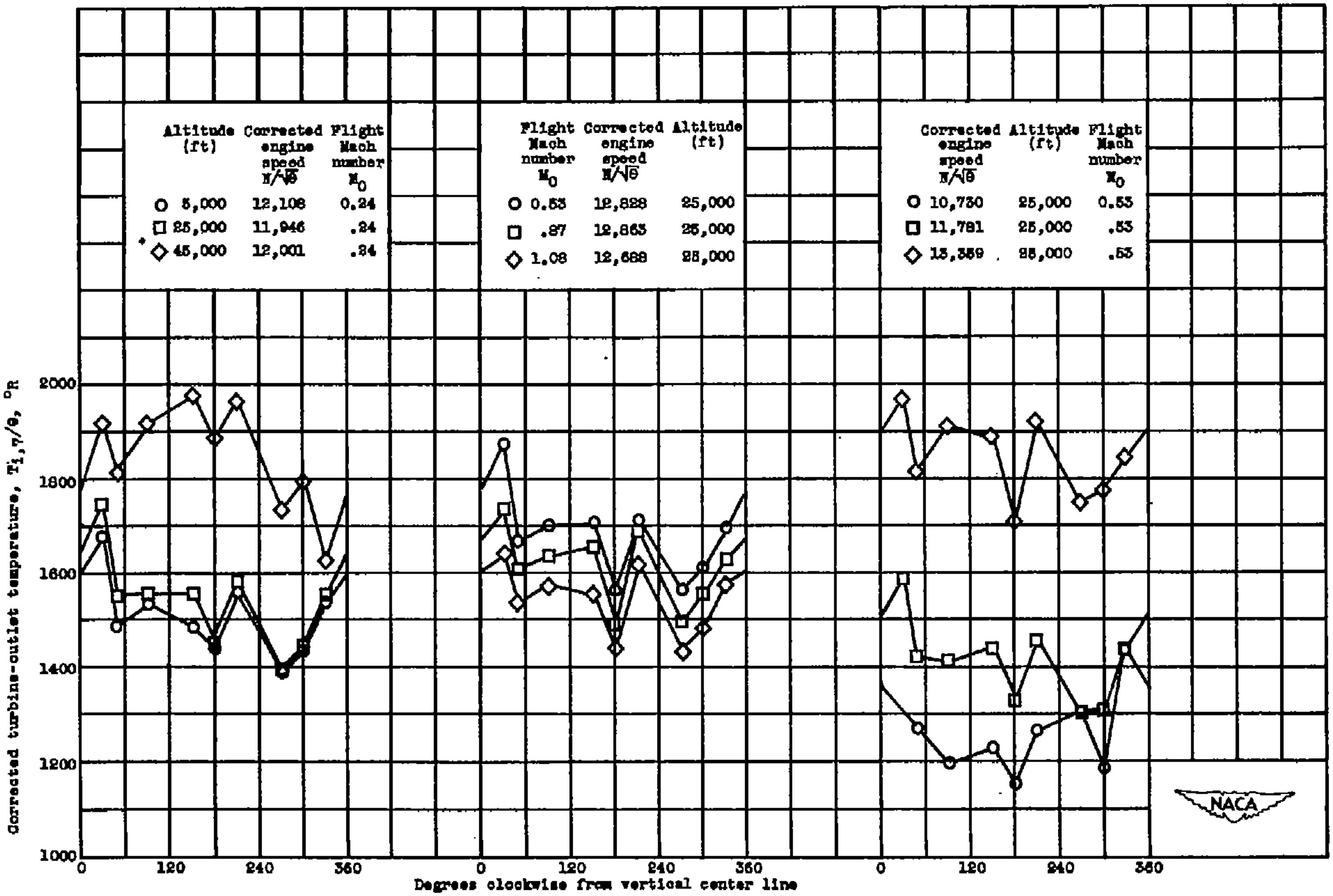
Figure 17. - Effect of altitude, flight Mach number, and engine speed on radial distribution of average corrected indicated temperature at turbine outlet, station 7.

[redacted]



(b) Modified engine.

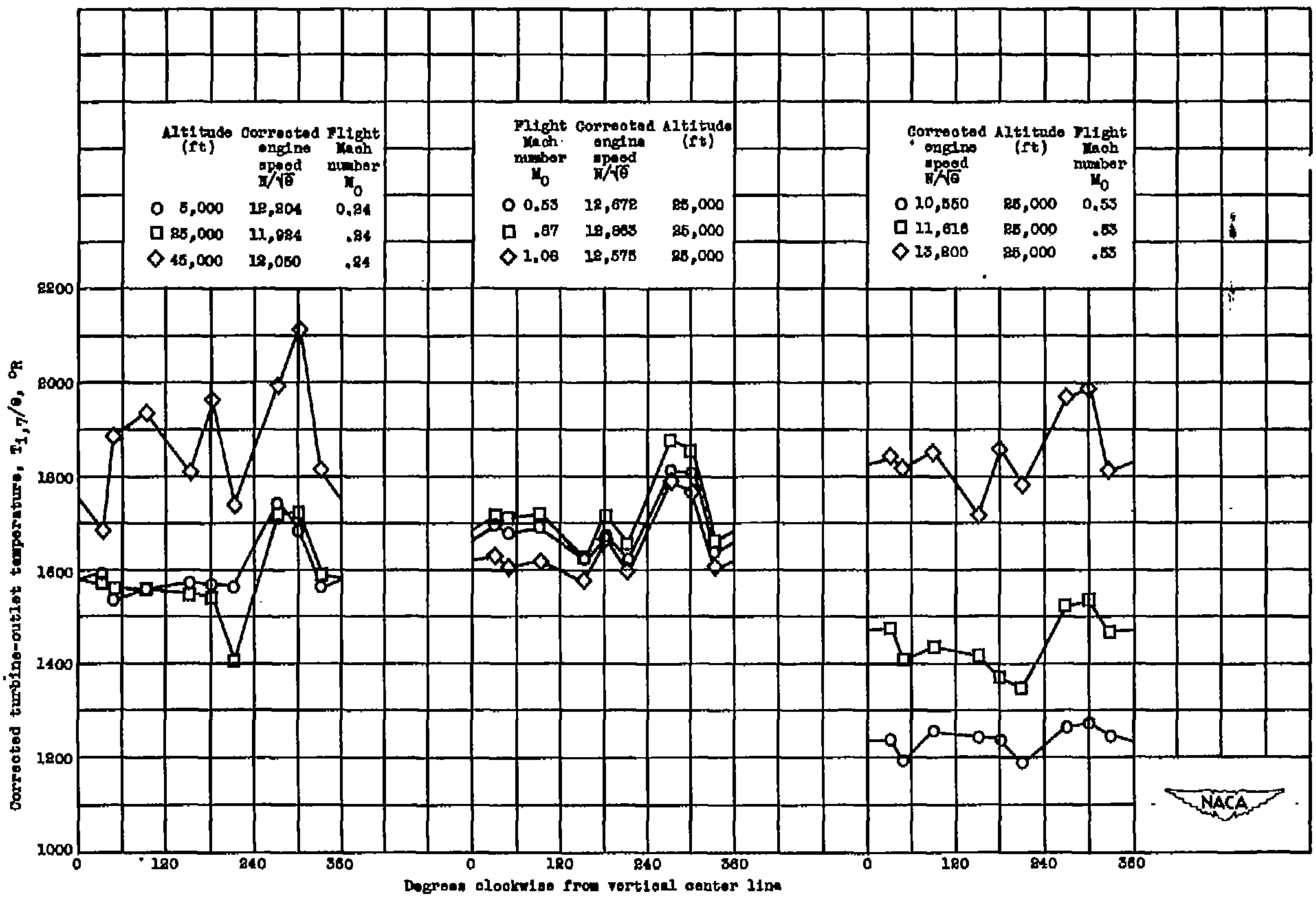
Figure 17. - Concluded. Effect of altitude, flight Mach number, and engine speed on radial distribution of average corrected indicated temperature at turbine outlet, station 7.



(a) Original engine.

Figure 18. - Effect of altitude, flight Mach number, and engine speed on circumferential distribution of corrected indicated temperature at turbine outlet, station 7, $1\frac{3}{4}$ inches from outer wall.

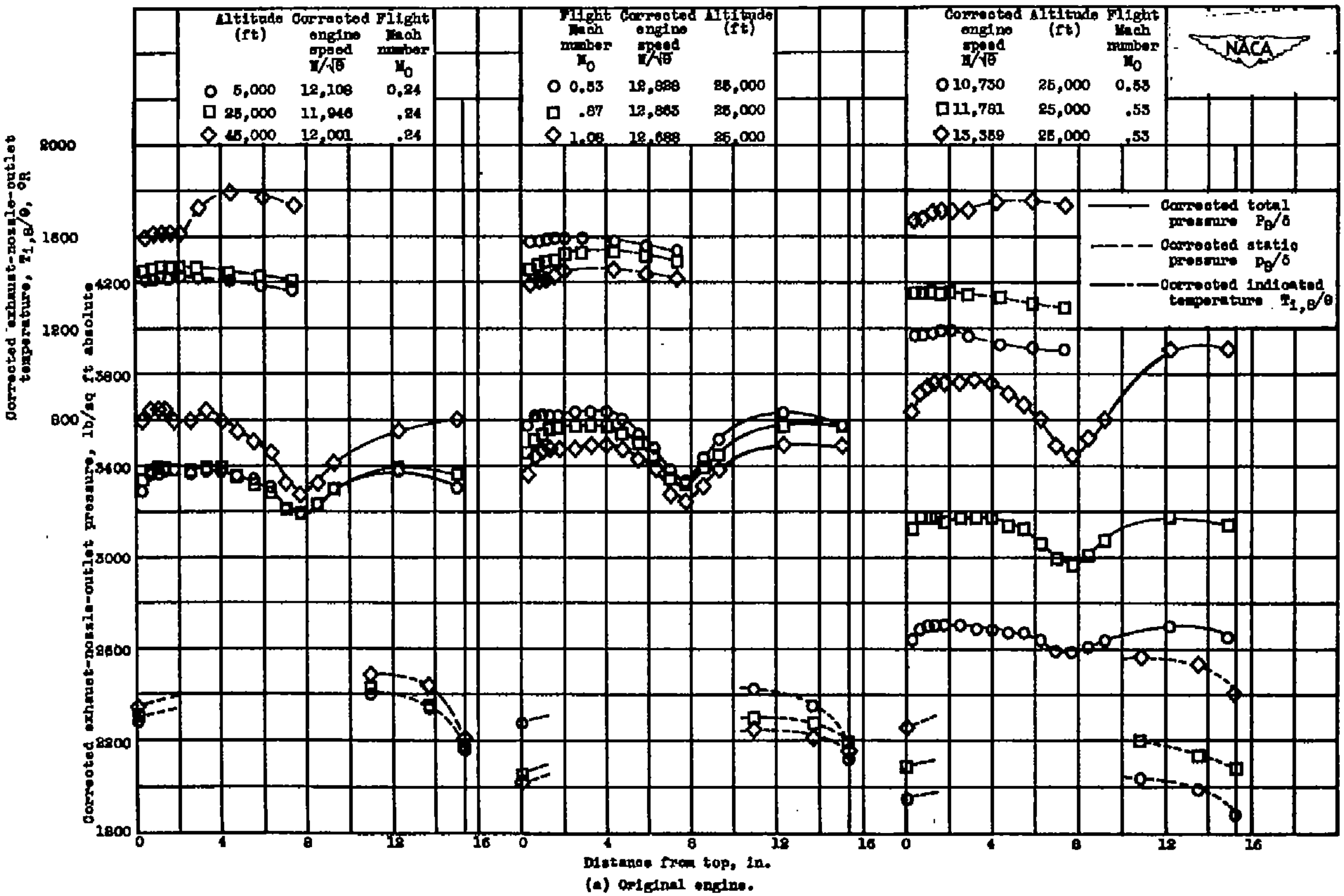
[redacted]



(b) Modified engine.

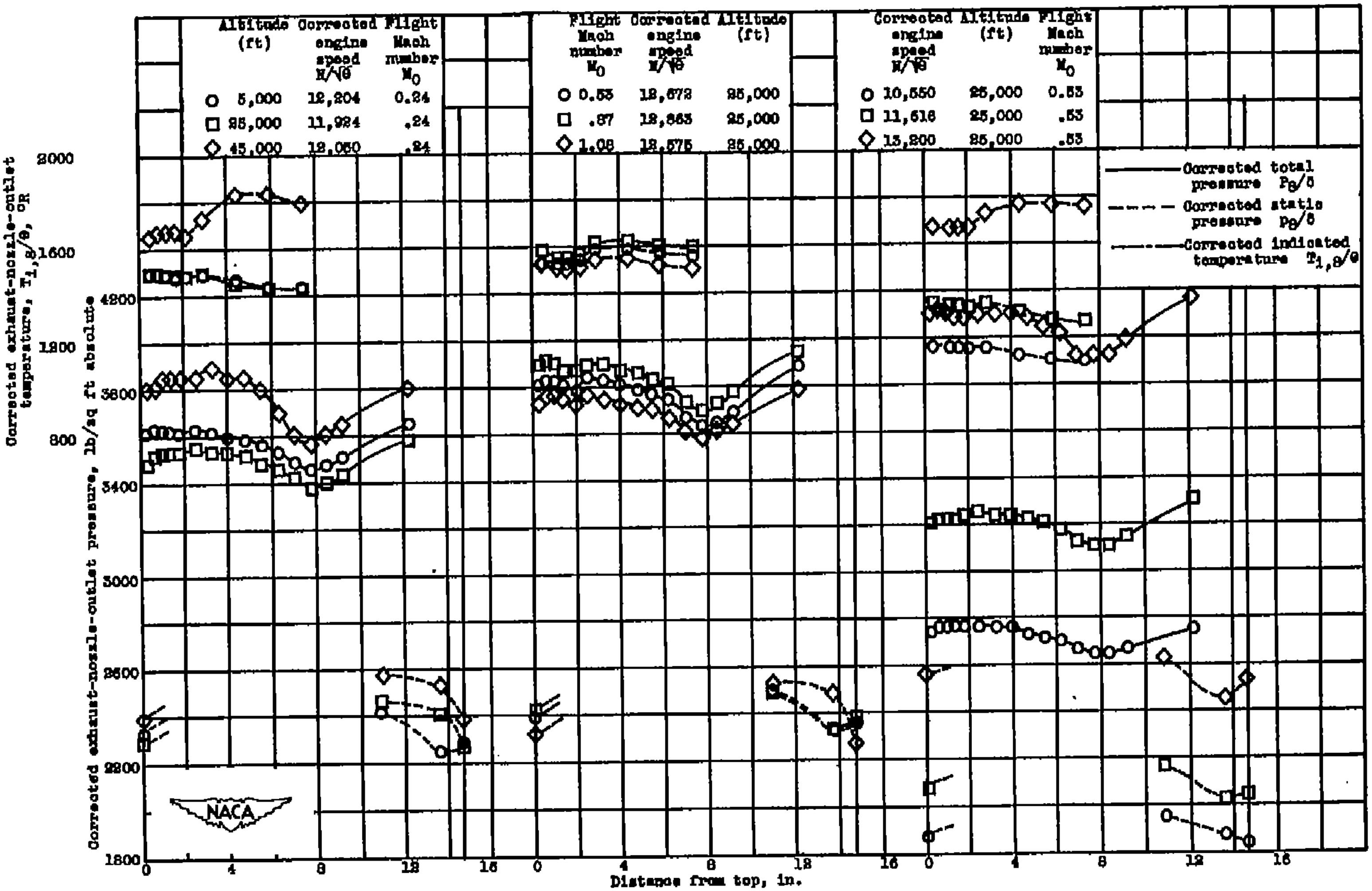
Figure 18. - Concluded. Effect of altitude, flight Mach number, and engine speed on circumferential distribution of corrected indicated temperature at turbine outlet, station 7, $1\frac{3}{4}$ inches from outer wall.

[redacted]



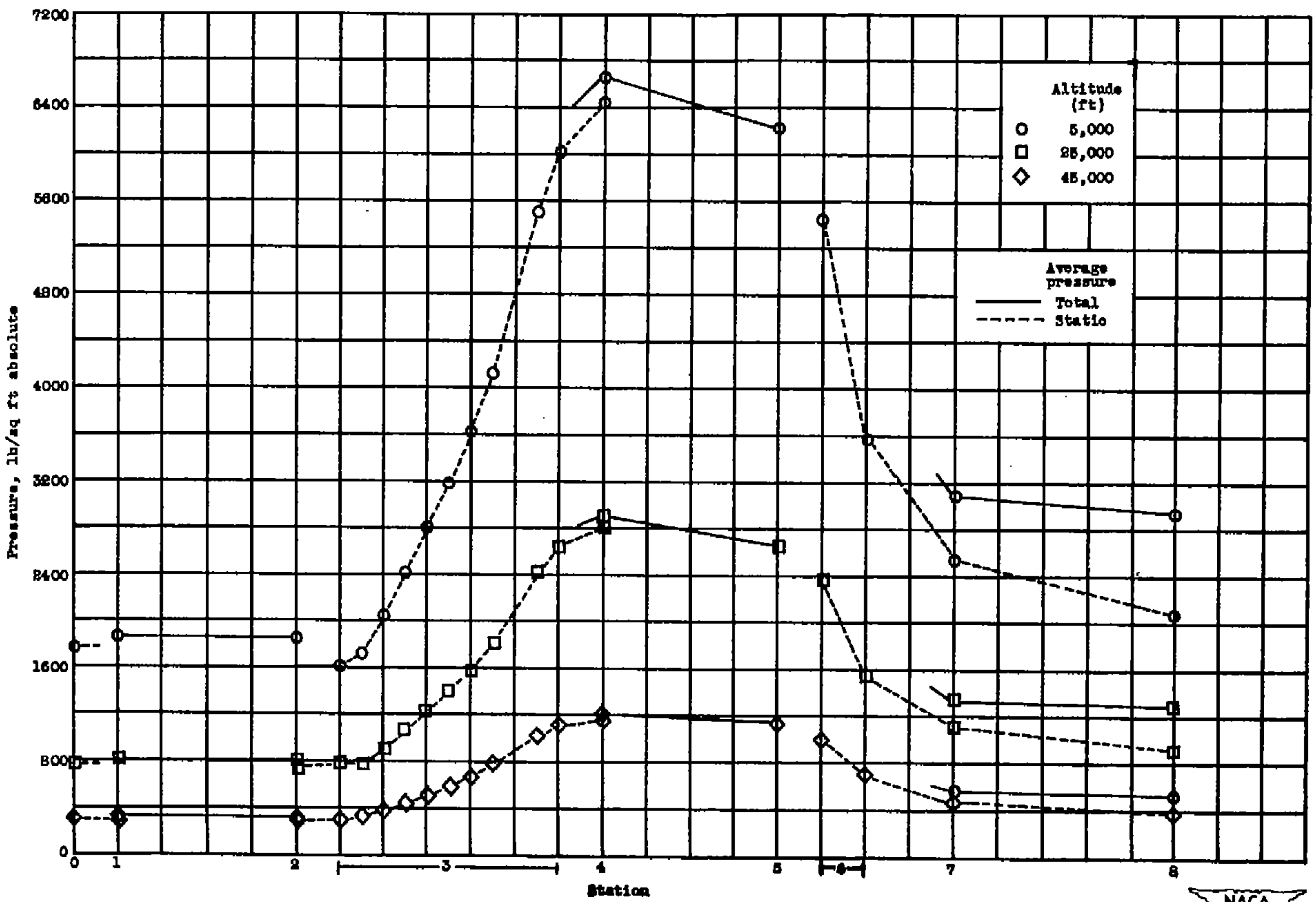
(a) Original engine.

Figure 19. - Effect of altitude, flight Mach number, and engine speed on distribution of corrected total pressure, static pressure, and indicated temperature at exhaust-nozzle outlet, station 8.



(b) Modified engine.

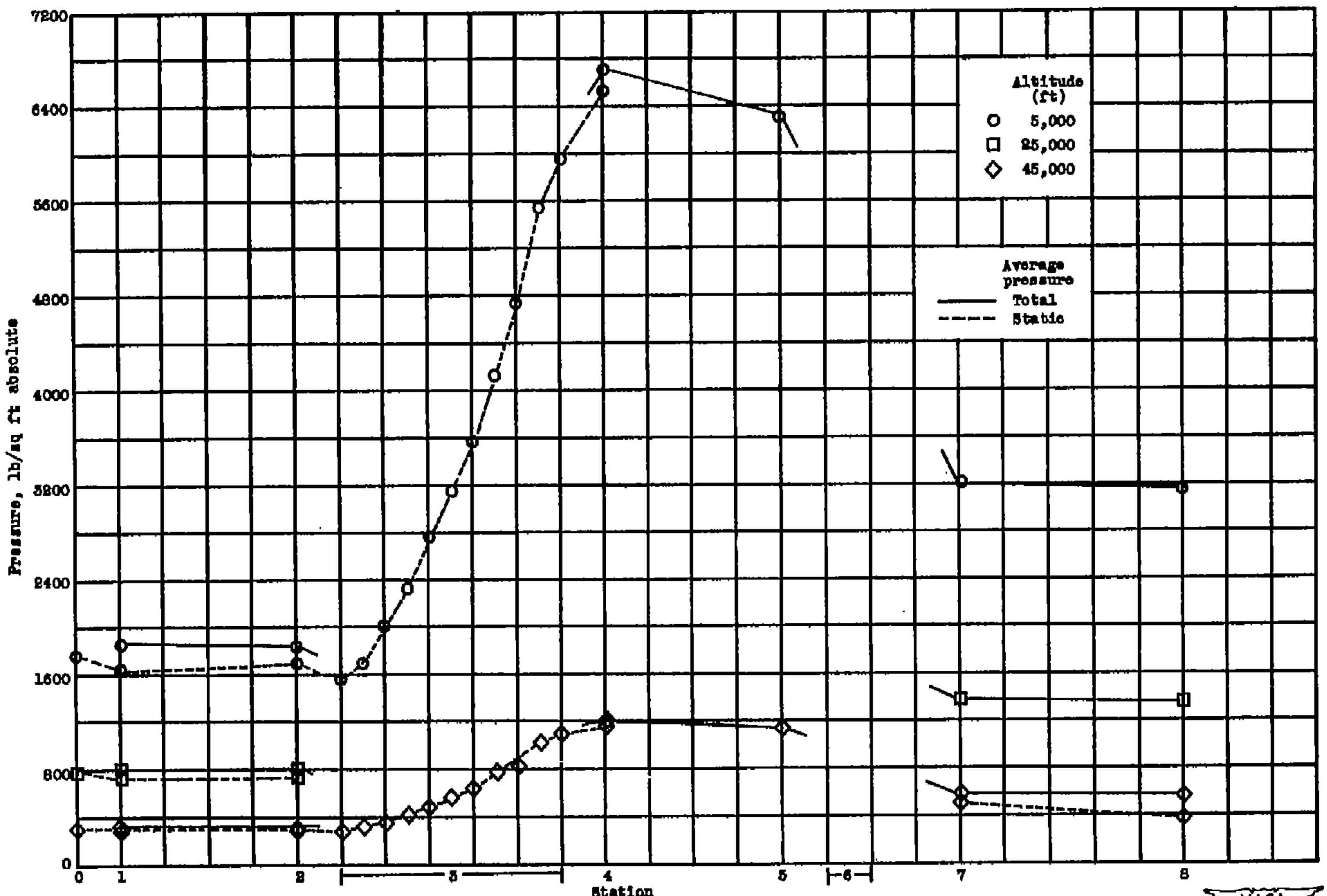
Figure 19. - Concluded. Effect of altitude, flight Mach number, and engine speed on distribution of corrected total pressure, static pressure, and indicated temperature at exhaust-nozzle outlet, station 8.



(a) Original engine.

Figure 20. - Variation of average total and static pressures through engine for several altitudes. Flight Mach number, 0.24; corrected engine speed, approximately 12,000 rpm.

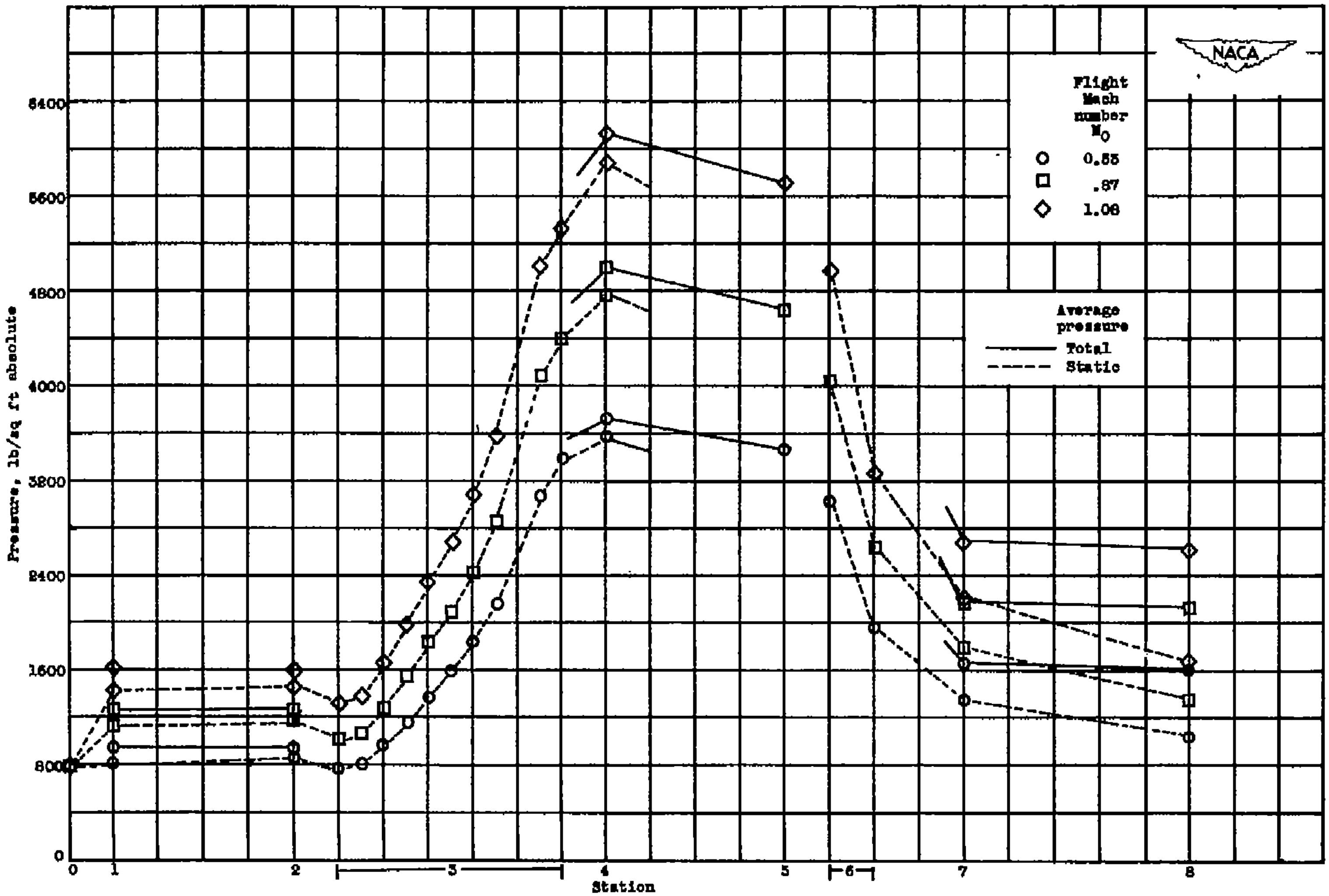
[redacted]



[redacted]

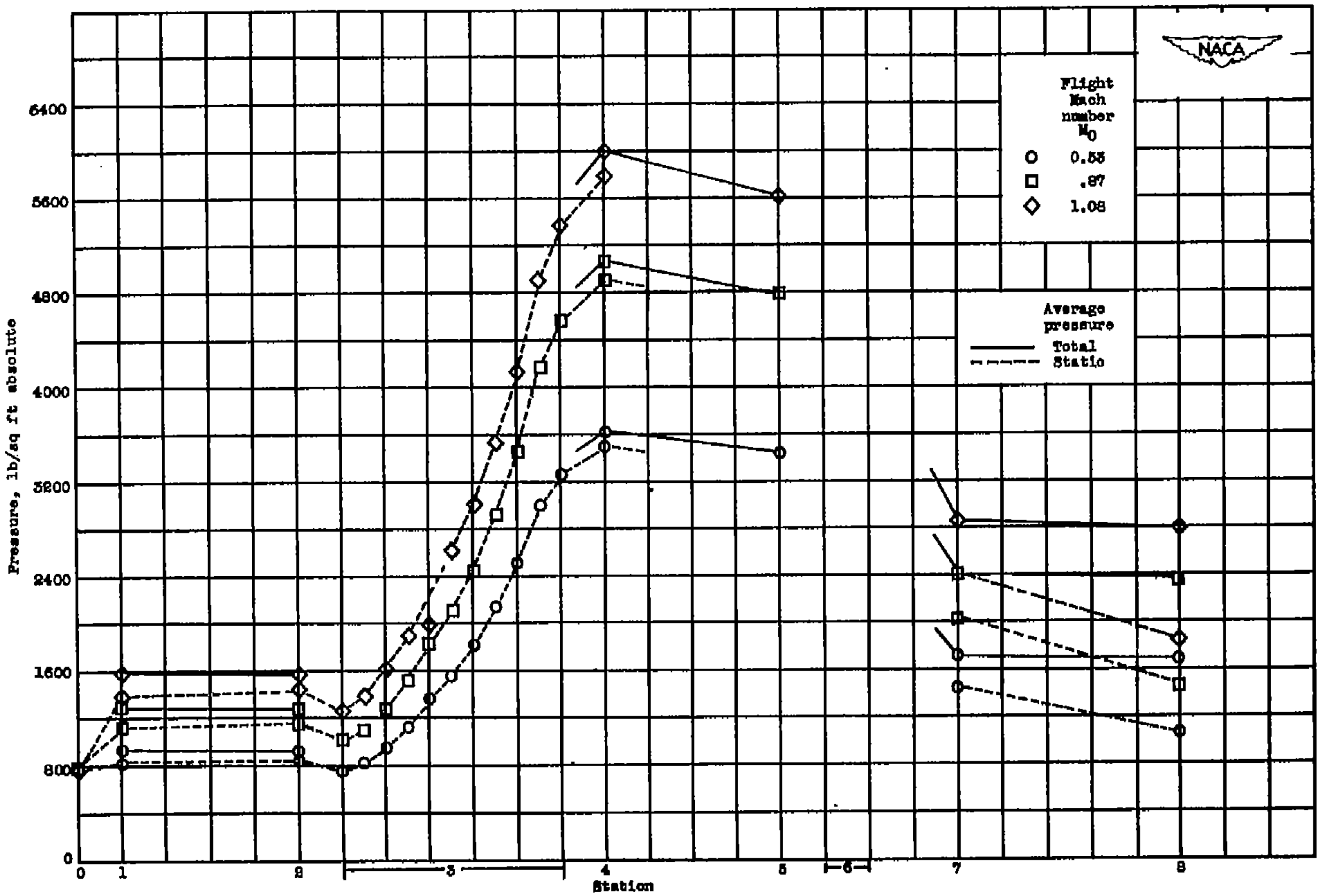
(b) Modified engine.

Figure 20. - Concluded. Variation of average total and static pressures through engine for several altitudes. Flight Mach number, 0.34; corrected engine speed, approximately 12,000 rpm.



(a) Original engine.

Figure 21. - Variation of average total and static pressures through engine for several flight Mach numbers. Altitude, 25,000 feet; corrected engine speed, approximately 12,700 rpm.

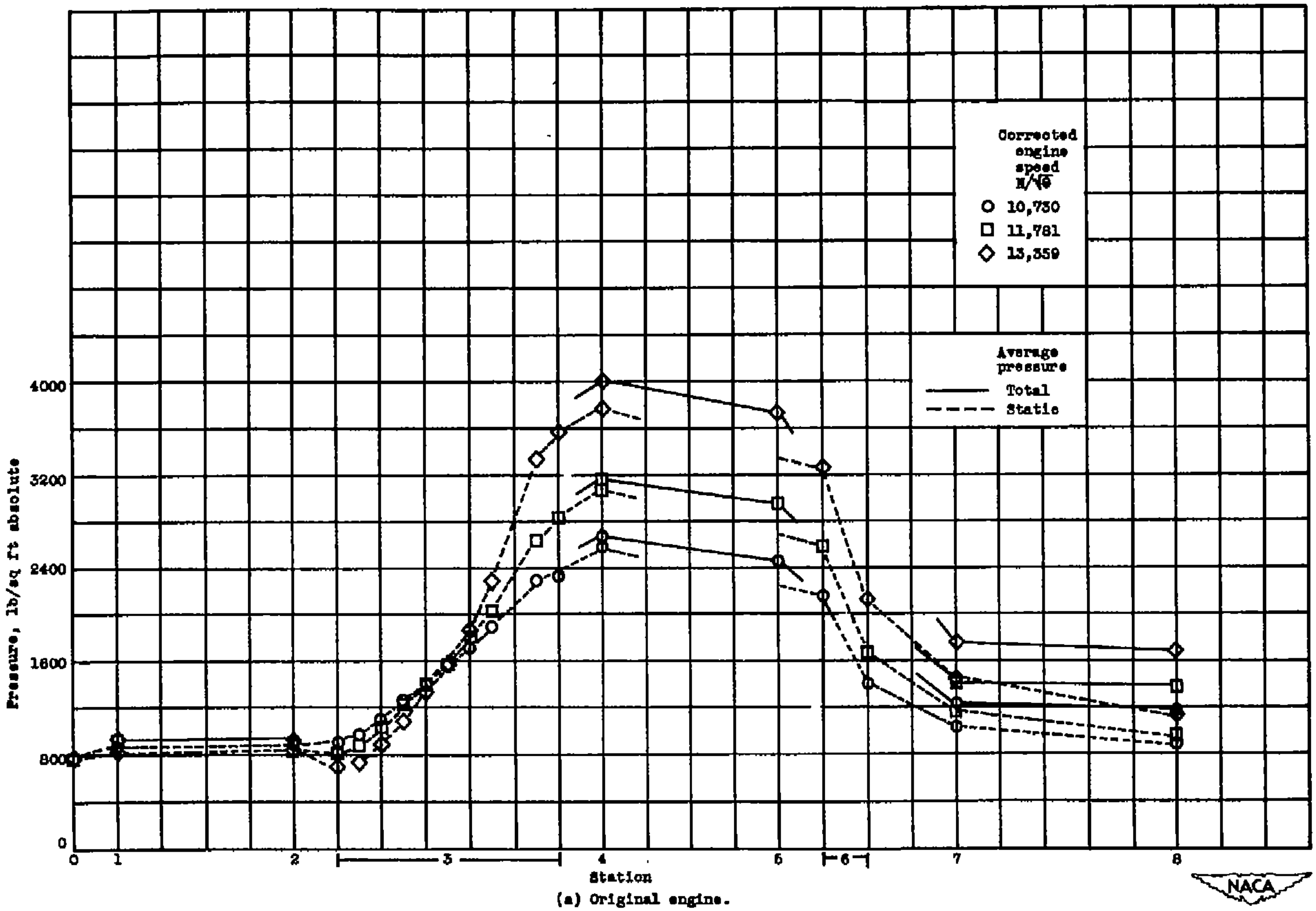


(b) Modified engine.

Figure 21. - Concluded. Variation of average total and static pressures through engine for several flight Mach numbers. Altitude, 25,000 feet; corrected engine speed, approximately 12,700 rpm.

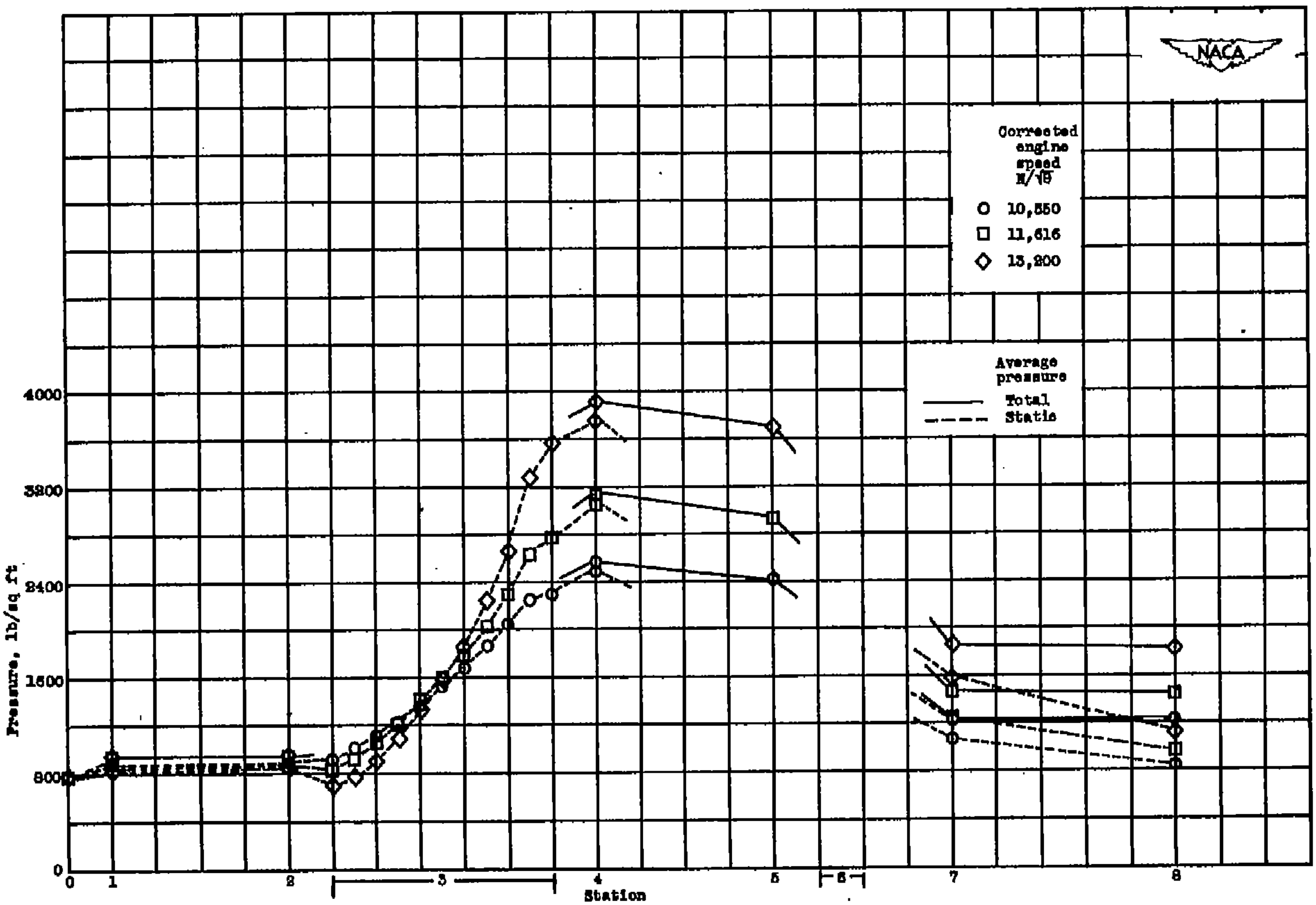
[redacted]

[redacted]



(a) Original engine.

Figure 22. - Variation of average total and static pressures through engine for several engine speeds. Altitude, 25,000 feet; flight Mach number, 0.57.



(b) Modified engine.

Figure 22. - Concluded. Variation of average total and static pressures through engine for several engine speeds. Altitude, 25,000 feet; flight Mach number, 0.53.